FR-19691-4
VOLUME I
1 DECEMBER 1989

SPACE TRANSPORTATION BOOSTER ENGINE CONFIGURATION STUDY

FINAL REPORT (DR4)

EXECUTIVE SUMMARY

31 MARCH 1989

CONTRACT NAS8-36857
MODIFICATION NO.10

Pratt & Whitney
Government Engine Business
P.O. Box 109600
West Palm Beach, Florida 33410-9600

Prepared for
Procurement Office
George C. Marshall Space Flight Center
National Aeronautics and Space Administration
Marshall Space Flight Center, AL 35812

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FOREWORD

This study was conducted by the Pratt & Whitney/Government Engine Business (P&W/GEB) of the United Technologies Corporation under NASA/MSFC contract NAS8-36857. The NASA/MSFC program manager was Mr. J. Thomson. The Pratt & Whitney program manager was Mr. W. A. Visek, Jr., and D. R. Connell was the booster engine program manager.

The technical effort started in March 1986 and was completed in March 1989. The study is presented in three volumes.

Volume I — Executive Summary
Volume II — Final Report
Volume III — Program Cost Estimates

Special thanks go to the numerous individuals at NASA, UTC, and the major vehicle contractors who contributed to this study effort.

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SECTION 1.0
INTRODUCTION

The United States is experiencing a critical need to place large payloads in low earth orbit. This need exceeds the capability of current and planned fleets of Titan IV and Space Shuttle launch vehicles, and reflects the requirements of the National Aeronautics and Space Administration (NASA), the U. S. Air Force, the Strategic Defense Initiative Organization (SDIO), and the civilian sector.

The Advanced Launch System (ALS) will provide a low cost, reliable means of satisfying this need. The ALS will enable the United States to meet defense, national, and civil launch requirements, while expending fewer resources on launch vehicles.

The objective of the Space Transportation Booster Engine Configuration Study is to contribute to the ALS development effort by providing highly reliable, low cost booster engine concepts for both expendable and reusable rocket engines.

An artist's concept of a fully reusable booster with a partially reusable core vehicle is shown in Figure 1-1.

The objectives of the Space Transportation Booster Engine (STBE) Configuration Study were: (1) to identify engine configurations which enhance vehicle performance and provide operational flexibility at low cost, and (2) to explore innovative approaches to the follow-on Full-Scale Development (FSD) phase for the STBE.

The Pratt & Whitney (P&W) overall technical approach to the study, shown in Figure 1-2, was based on the STBE technical requirements and guidelines presented in the Statement of Work (SOW). These requirements and guidelines were modified continually as the results of the joint NASA/Air Force Space Transportation Architecture Study (STAS), and later the Advanced Launch System (ALS), became available. As a result, the study effort was completely supportive of and interactive with the ALS and other launch vehicle studies. The schedule of the STBE Phase A, including the three extensions and the interim final reporting documentation, is shown in Figure 1-3.

The STBE Configuration Study consisted of six tasks. Task I (SOW Task 5.1) consisted of parametric analyses and trade studies. First, the system design requirements and features were defined, and the information base was established. Second, the STBE configurations that enhance performance and provide operational flexibility at low cost were identified, and the requirements for those engine configurations for the projected missions were defined.

During Task II (SOW Task 5.2), P&W developed a plan to evaluate the STBE configurations identified in Task I and established criteria to select the most promising configurations. The Configuration Evaluation and Criteria Plan used overall system life cycle costs as the figure of merit and included considerations of mission and vehicle requirements, operational flexibility, schedules (along with their risks), required technological advances, and facility requirements. The evaluation and selection criteria were compatible with the NASA requirements and the STAS results.

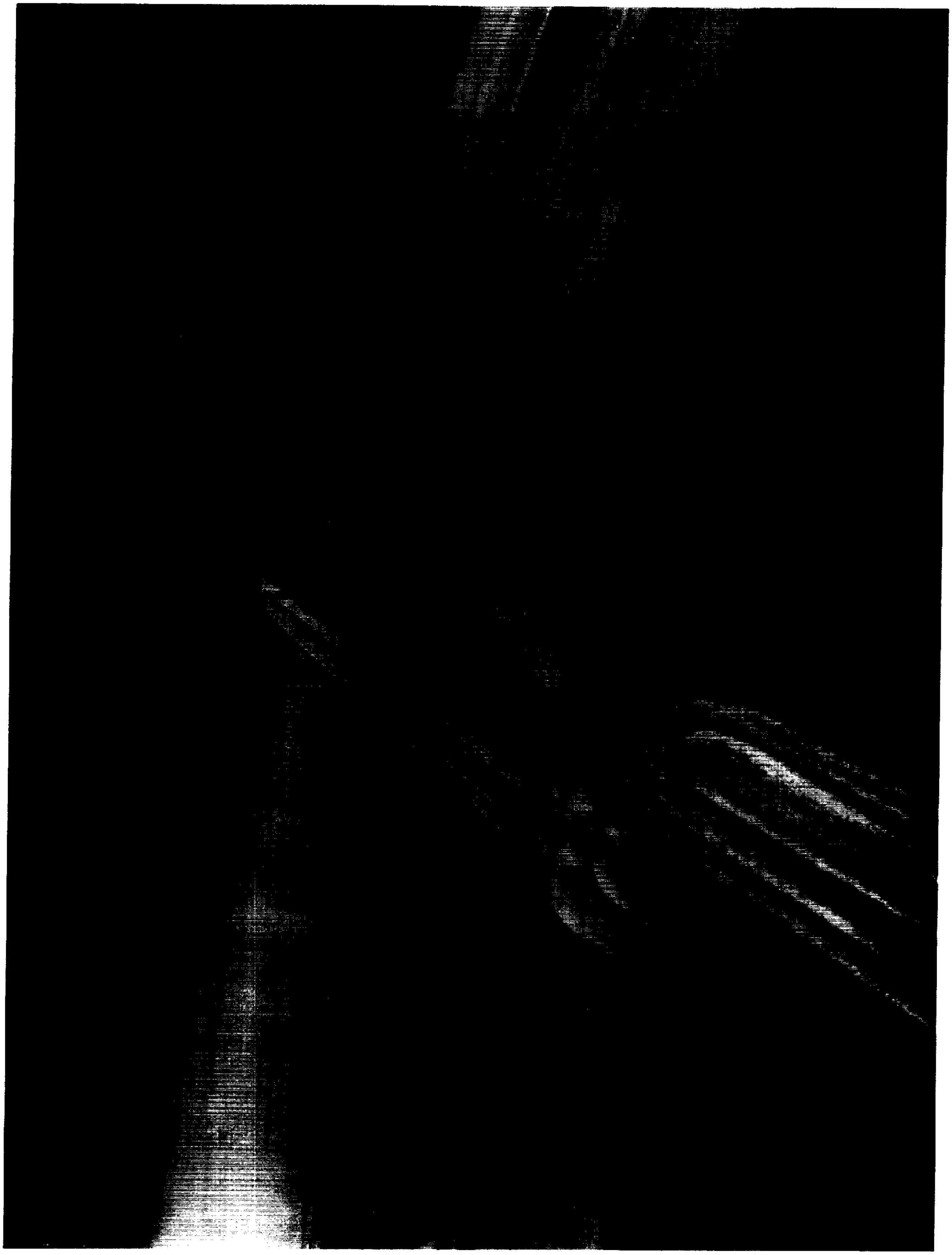

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Figure 1-1. Artist's Concept — Fully Reusable Booster With Partially Reusable Core Vehicle

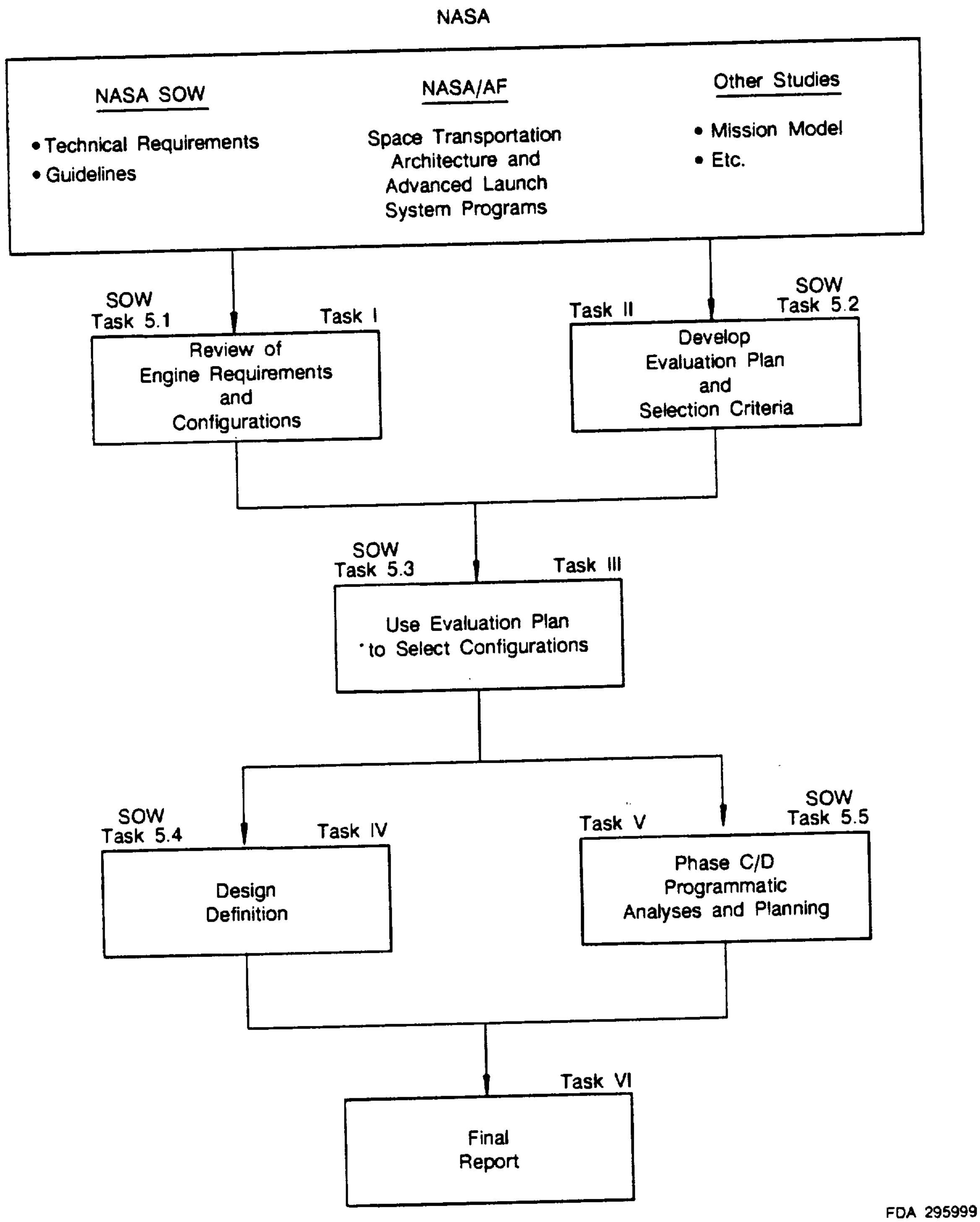

Figure 1-2. Overall Approach to Space Transportation Booster Engine Configuration Study

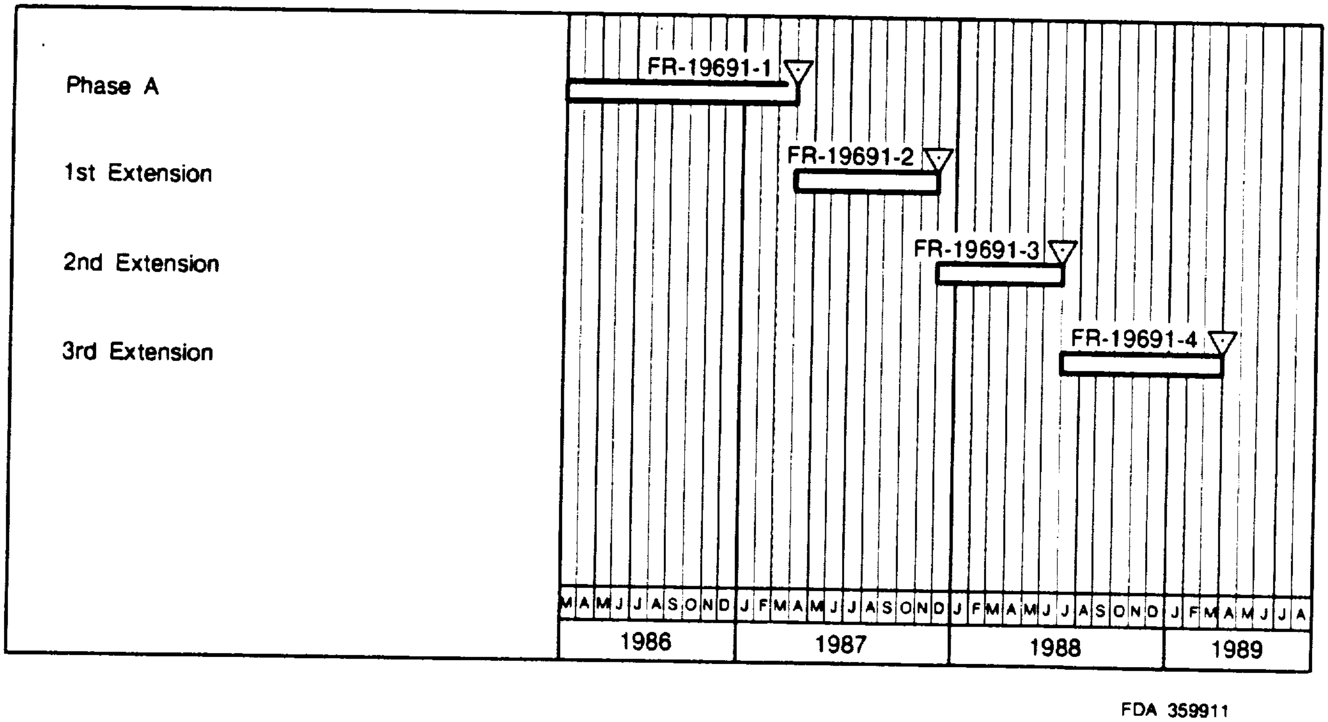

Figure 1-3. STBE Phase A and Extensions

During Task III (SOW Task 5.3), P&W assessed the STBE configurations and requirements identified during Task I using the Configuration Evaluation and Criteria Plan developed during Task II. This process, based on minimizing life cycle cost (LCC), was used to select the most promising engine candidate as agreed to by NASA and P&W.

The selected engine candidate was then the subject of Tasks IV and V. During Task IV (SOW Task 5.4), P&W completed the conceptual designs of the selected candidate. Under this task, P&W prepared the Design Definition Document (DR8), including a preliminary Interface Control Document (ICD) and preliminary Contract End Item (CEI) Specification. Task V (SOW Task 5.5) was conducted concurrently with Task IV and provided the plans for FSD. These plans included schedules, facility requirements, a Work Breakdown Structure (WBS) and dictionary, a cost analysis, and an Environmental Impact Analysis (DR10).

During Task VI, all of the technical reviews, status reports, and the final report were prepared.

The Interim Preliminary Reports were published at the milestones shown in Figure 3. The information and studies reported within these documents are referenced but not repeated in this Final Report.

Volume II of this Final Report contains all of the work conducted under the contract during the time period July 1, 1988 to March 31,1989. Section 2.0 of this volume, Evolution of STBE Phase A, provides a narrative of the STBE Phase A effort that ties together the reference documents.

All costs contained in this volume are engineering estimates. These costs should not be considered as contractual commitments and should be used for Life Cycle Cost (LCC) evaluations and planning purposes only.

The STBE Program WBS and cost estimates are presented in Volume III.

SECTION 2.0
EVOLUTION OF STBE DURING PHASE A

The Space Transportation Booster Engine (STBE) configuration study evolved over the three-year contract period. A brief overview of the significant phases of the study is shown in Figure 2-1.

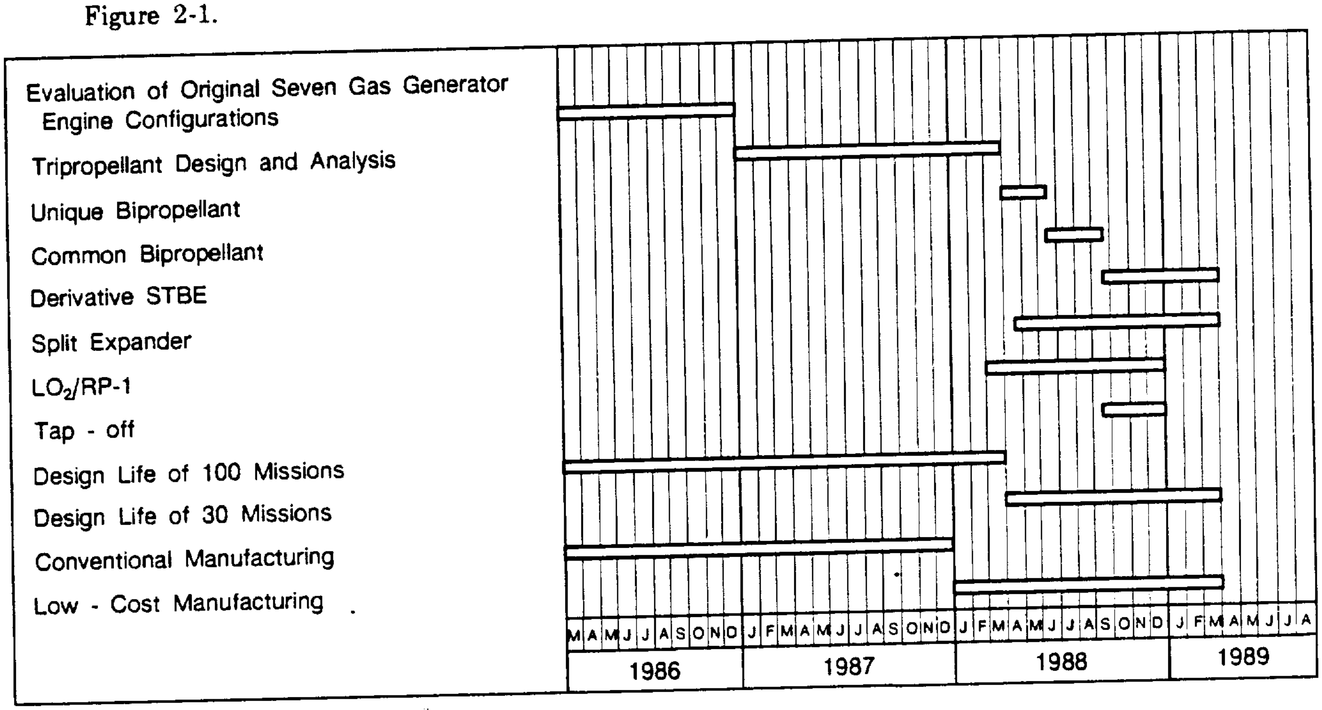

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Figure 2-1. STBE Study Significant Phases

Seven Gas Generator engine configurations were initially identified that met the requirements set forth in Task 1, Vol. II of FR-19691-1. Their characteristics are given in Table 2-1. These configurations were assessed using the Configuration Evaluation and Criteria Plan developed during Task II. The engine evaluation process was based on determining the total life cycle cost (LCC) of a launch system using the ground rules for the trajectory, the vehicle, and for the programmatic considerations. In recent years, LCC has become the accepted standard criteria on which to make the "best" choice because it includes all the important elements of engine evaluation criteria: performance, weight, development difficulty, risk, and operations as well as cost. LCC is the figure of merit which encompasses the total system, and therefore requires system level analysis.

Figure 2-2 shows an overview of the launch vehicle/rocket engine optimization procedure that was used as the basis for the present study. After the study ground rules were established, the matrix of design variable (parameter) combinations was selected. Engine performance and weight were then calculated for each of the variable combinations. The vehicle characteristics were obtained by an iterative procedure that loops through the Vehicle Weight and Sizing Program, the Trajectory Program interface, and the engine performance and weight data, until a converged mission-capable vehicle was defined. The characteristics of this vehicle were then passed on to the Vehicle LCC Program, which also receives input from the Engine LCC Program. For each vehicle in the parametric matrix, LCC and weight data are passed into the Regression

Program which fits a multivariable surface defining LCC as a function of the design variables. The Optimizer Program then interrogates the surface and searches for the combination of design variables which results in a minimum LCC vehicle.

Table 2-1. *STBE Candidate Engine Configurations — All Gas Generator Cycles*

	STBE-1A	*STBE-1B*	*STBE-2*	*STBE-3*	*STBE-4*	*STBE-5*	*STBE-6*
Propellants	LO_2/RP-1	LO_2/RP-1	LO_2/RP-1	LO_2/CH_4	LO_2/CH_4	LO_2/C_3H_8	LO_2/C_3H_8
Coolant	RP-1	LO_2	LH_2	CH_4	LH_2	C_3H_8	LH_2
Mixture Ratio	2.90	2.90	3.12	3.57	3.64	3.20	3.38
Chamber Press (psia)	1275	1667	3500	2333	3500	2333	3500
Thrust							
Vacuum (lbf)	736,100	735,900	706,000	713,100	705,800	715,100	705,800
Sea Level (lbf)	625,000	625,000	625,000	625,000	625,000	625,000	625,000
Specific Impulse							
Vacuum (sec)	316.0	318.4	360.1	341.5	369.5	333.9	363.2
Sea Level (sec)	264.3	273.5	318.2	302.6	326.5	291.4	321.0
Area Ratio	25	35	55	40	55	40	55
Length (in.)	152	155	143	143	143	143	143
Diameter (in.)	98	98	84	88	84	88	84
Weight (lb)	6750	6745	6925	6655	6845	6650	6885

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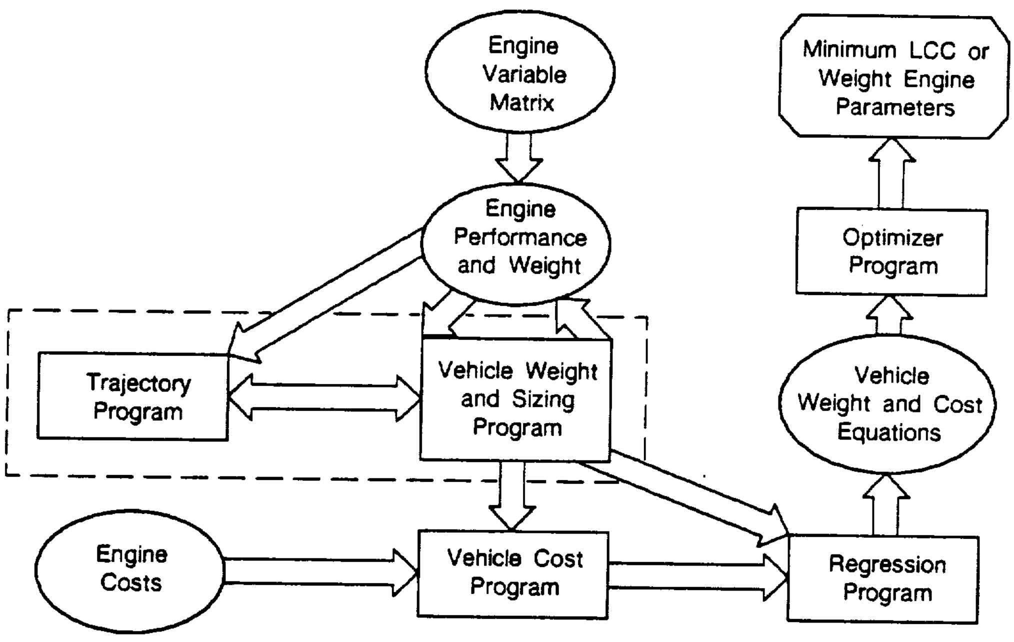
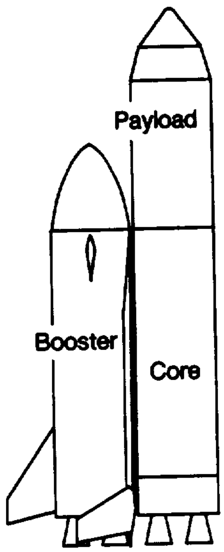


Figure 2-2. Launch Vehicle/Rocket Engine Optimization Procedure

The ground rules of this evaluation procedure were established jointly by Pratt & Whitney and the NASA Program Manager. Figure 2-3 describes the Launch Vehicle used in the analysis. A glideback booster with a 3000 ft/sec relative burnout velocity and a flyback booster with a 6000 ft/sec burnout velocity were evaluated to see if the optimum STBE characteristics changed. The trajectory ground rules are presented in Table 2-2, and the programmatic ground rules are presented in Table 2-3.



Payload	• 150,000 lb
Booster	• Advanced Hydrocarbon Propellants Fully Reusable Burnout Velocity 3,000 ft/sec Glideback 6,000 ft/sec Flyback
Core	• Advanced O_2/H_2 STME Partial Reusable (Recoverable P/A) Parallel Burn With Crossfeed
Variables	• Pc,O/F, ϵ } Each Stage

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Figure 2-3. STBE Study — Launch Vehicle Used in Evaluation Analysis

Table 2-2. *STBE Study — Trajectory Ground Rules*

Trajectory Ground Rules Were as Follows:

- East Launch/28.5° Inclination
- Powered Ascent to 75 × 150 nm Orbit
- Circularize at Apogee Using OMS
- Maximum Q < 1100 lb/sq ft
- Maximum G < 4.0 (Axial)
- Optimized Pitch Schedule

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Table 2-3. *STBE Study — Programmatic Ground Rules*

	Flyback Booster	*Core Vehicle*
Active Number Vehicles	8	8
Avg Launch/Year/Vehicle	6	6
Development Time		
— Engine	7 yr	7 yr
— Vehicle	6 yr	6 yr
First Flight	1995	1995
Operating Period	15 yr	15 yr
Vehicle Useful Life	100 Missions	100 Missions

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The detailed discussion of this assessment and the results are presented in Volume II of FR-19691-1. The engine configuration selected by this process to have the lowest life cycle cost was the LO_2/methane/hydrogen Tripropellant Gas Generator.

The following factors that make the LO_2/methane/hydrogen tripropellant the lowest life cycle cost engine configuration also make good engineering sense:

- *Combustion Stability* — Methane has the best history of combustion stability of all of the hydrocarbon rocket fuels. Also, the addition of hydrogen into the main combustion process will increase the burning rate. This increase in burning rate is predicted to make the combustion process even more stable.

- *Combustion Efficiency* — Although high combustion efficiencies have been obtained in LO_2/methane combustion systems, adding hydrogen to the combustion process increases the calculated combustion efficiency for the various hydrocarbon fuels for a resultant higher efficiency than LO_2/methane.

- *Cooling Capability* — The excellent cooling capability of liquid hydrogen has been established in several operational engine designs.

- *Ignition* — An oxygen/hydrogen torch igniter can be used. The ignition limits of oxygen and hydrogen are very broad. This allows ignition at low pressures and mixture ratios well away from the stoichiometric mixture ratio. The hydrogen/oxygen ignition source also heats the methane/oxygen mixture for easier main chamber ignition. The main chamber could also be started with only oxygen and hydrogen.

- *Cleanliness of Turbine Drive Gas* — The exhaust of the gas generator driving the turbines is hydrogen and steam; it is clean, and is used successfully in the Space Shuttle Main Engine.

- *Chamber Material Compatibility* — Hydrogen is known to be compatible with the copper alloys used in the design of combustion chambers. However, because of hydrogen embrittlement the usual care must be taken in the selection of materials.

- *Safety* — Both methane and hydrogen are lighter than air at ambient pressure and temperature, therefore, leaks or spills will not accumulate in low areas.

In summary, the selection of the LO_2/methane fueled, hydrogen cooled tripropellant engine configuration either eliminates or greatly reduces the risks associated with the design of high pressure, reusable hydrocarbon booster engines.

This tripropellant engine configuration selection was then carried into Tasks IV and V. During Task IV the conceptual design was completed. Concurrently, the plans for the Full Scale Development program were prepared. The detailed results of both of these efforts are presented in Volume II of FR-19691-1.

Study efforts on this tripropellent configuration continued through the first extension of the Phase A contract. The results were given in Sections 2, 4 and 5 of FR-19691-2. Also during this period, vehicle studies produced information on bipropellant booster engines. P&W began work to define the characteristics of a bipropellant booster engine, and this work is documented in section 3 of FR-19691-2.

At this point in the conceptual design process, P&W's Manufacturing Division studies revealed innovative low-cost design concepts and manufacturing techniques for the STBE configuration.

The concurrent studies by the ALS vehicle contractors were now showing some results. These results showed that a bipropellant gas generator engine cycle is more cost effective than the tripropellant. Also, at about this same time, NASA changed the engine life requirement from 100 missions to 30 missions, a number thought to be more realistic. The tripropellant configuration was then set aside and the effort was focused upon the LO_2/methane bipropellant gas generator. This configuration had the second lowest life cycle cost after the original tripropellant selection in the evaluation. During the second extension of the Phase A contract, P&W completed its studies on the tripropellant engines and continued working on the design characteristics and configurations of several bipropellant engines, including the LO_2/methane gas generator engine. The results of these studies, including the work performed on the tripropellant engine during that extension period, is given in FR-19691-3. The reasons why methane was consistently better than either propane or RP-1 are given in Table 2-4.

Table 2-4. *Methane Advantages Over Propane and RP-1*

• Highest Combustion Efficiency	• Very Stable Combustion
• More Predictable Heat Flux	• A Good Coolant, with High Coking Temperatures
• Clean Gas Generator Gas	• Allows Transpiration Cooling
• Simplifies Injector Design	• Allows Coaxial Injection of Gaseous Fuel
• Self Purging Reduces Cleaning Requirements	• Improves the Injector Face Cooling
	• Reduces environmental impact of spills. The volatile, non-toxic gas readily disperses.

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The conceptual design of the LO_2/methane bipropellant gas generator engine is presented in detail in paragraph 3.1.1. The complete plans for its full-scale development have been prepared and are presented in Section 5.0 of Volume II.

In late 1987, in-house studies by Pratt & Whitney started to show that a split expander engine cycle would be more cost effective than a gas generator cycle. The split expander cycle differs from the standard expander cycle used in the RL10 engine by separating a portion of the fuel flow at the first-stage pump and directing that flow directly to the injector. The remainder of the fuel flow completes the standard expander cycle. The total heat "pickup" in the nozzle by this flow is approximately the same as a standard expander cycle. This flow cools the chamber and drives the turbines. Since flow and temperature trade proportionally in turbine power, the split expander low flowrate at the higher temperature will provide the same turbine power. The pump work will be reduced due to the reduction in flow through the second stage pump. This reduced power requirement provides the capability for a higher chamber pressure. Furthermore, the increase in fuel temperature at the turbine inlet ensures that gaseous fuel will be maintained (throughout the turbine) at high thrust level conditions. With the approval of NASA, the analysis and evaluation of the split expander engine cycle became part of this contract effort. The analysis of the split expander continued through the remainder of the contract at a lower level of effort than the gas generator cycle. The details of the early design analyses of the split expander cycle engines can be found in paragraphs 3.5.1 and 3.6.1.

In 1988 the ALS Vehicle Contractor studies began to show some advantage to having a common engine for both the booster and the core vehicles, i.e., one engine that could meet both the requirements of the STME when operated with hydrogen/oxygen propellant, and the requirements of the STBE when operated with LO_2/methane propellants. The design analyses of such an engine, presented in paragraph 3.3.1, showed that the design was possible, but penalized the hydrogen/oxygen core vehicle engines because of the required additional weight.

At about this time it became more evident that the immediate need was for a LO_2/hydrogen engine (STME). The LO_2/methane engine requirement was slipping toward the end of the ALS program life.

The STBE contract emphasis then finally shifted to a LO_2/methane booster engine that could be obtained by modifying the STME engine design and still attain a sea level thrust of 600,000 lbf or greater. This requirement was met by both the gas generator engine cycle and the split expander engine cycle. This engine is known as a Derivative STBE.

SECTION 3.0
DESCRIPTION OF ENGINES

In the period 1 July 1988 through 31 March 1989, conceptual designs of seven engine configurations were completed. This section presents a brief description of these designs.

The designs include:

- Derivative LO_2/CH_4 Gas Generator Cycle Engine
- Unique LO_2/CH_4 Gas Generator Cycle Engine
- Common LO_2/CH_4 Gas Generator Cycle Engine
- Unique LO_2/RP-1 Gas Generator Cycle Engine
- Derivative LO_2/CH_4 Split Expander Cycle Engine
- Unique LO_2/CH_4 Split Expander Cycle Engine
- Unique LO_2/CH_4 Tap-Off Cycle Engine

For brevity and to minimize repetition, only the description of the derivative LO_2/CH_4 Gas Generator Engine includes the engine components. Only the overall engine characteristics are described for the remaining six engines.

3.1 DERIVATIVE STBE LO_2/CH_4 GAS GENERATOR CYCLE ENGINE

3.1.1. Engine Design Evolution

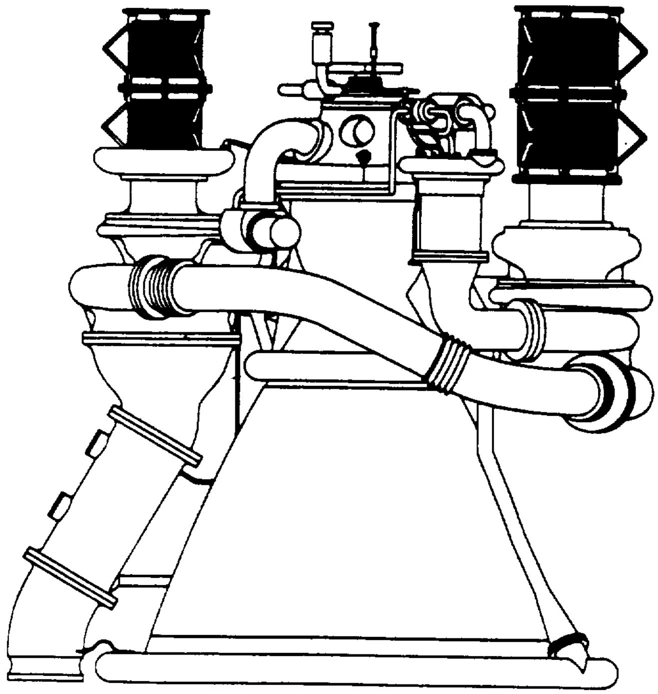
The Derivative STBE gas generator cycle engine concept began as a result of the common engine studies. In addition, the need for a unique engine, optimized for core vehicle use, ruled out the possibility of funding a separate, unique booster engine design as well. As discussed in paragraph 3.3.1, the common engine designs consisted of a common O_2/H_2 gas generator (GG) cycle engine that had slightly reduced performance characteristics than the unique STME and a 644K common LO_2/CH_4 Gas Generator Cycle engine that had reduced thrust compared to the 750K unique STBE. Although hardware commonality between the two engines was maximized, the concept proved to be unacceptable when the following ground rules were established:

1. No performance, cost, or weight penalties of the unique STME engine design are permitted

2. The STBE engine will use as much of the unique STME hardware as possible, and thus will be a derivative of the STME

3. The booster engine application must obtain 600K sea level thrust or greater.

The conceptual design that arose as a result of this study is known as the Derivative STBE; it is a derivative of the LO_2/LH_2 STME, but uses LO_2/CH_4 and is designed for booster applications. Figure 3.1.1-1 presents an engine assembly drawing and the overall engine characteristics. This derivative engine is the current baseline design for the STBE, therefore, the parametric equations, the ICDs, and CEI documents included in Volume II this report pertain to the derivative engine.



Propellants	CH_4/LO_2
Mixture Ratio	2.7
Chamber Pressure	2,250 psia
Thrust - Vacuum	711,823 lb
- Sea Level	644,898 lb
Specific Impulse - Vacuum	328.4 sec
- Sea Level	297.5 sec
Nozzle Area Ratio	28
Diameter	91 in.
Length	99 in.
Weight	6,960 lb

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Figure 3.1.1-1. STBE Derivative Gas Generator Engine Design Conditions

3.1.2 Engine Cycle

The STBE derivative is a LO_2/CH_4 gas generator cycle adapted from the STME LO_2/LH_2 GG cycle engine. The STBE operates at a main chamber pressure of 2250 psia with a sea level thrust of 645K lbf. The nozzle area ratio for this engine is 28:1 and delivers a sea level specific impulse of 297.5 seconds. Figure 3.1.1-1 presents selected engine characteristics at the fixed operating conditions.

Components of the STBE derivative that will be common with the STME are the main injector, gas generator, tubular nozzle, engine controller, igniters, GO_2 heat exchanger (HEX), POGO suppressor, instrumentation, vehicle interfaces, and 80 percent of the ducting. Items that will be redesigned for the STBE derivative are the combustion chamber, oxidizer pump, oxidizer turbine, fuel turbine, GG oxidizer valve, GG fuel valve, and the gimbal. Table 3.1.1-1 summarizes the common hardware components between the STME and Derivative STBE gas generator engines.

3.1.2.1 Flowpath Description

A simplified flow schematic for the STBE derivative is presented in Figure 3.1.2-1 showing the major components and flowpaths. Liquid methane and liquid oxygen enter the engine at a NPSH level, supplied by the vehicle, sufficient for the high-speed, high-pressure pumps, with no boost pumps required.

Table 3.1.1-1. STME and Derivative STBE Gas Generator Engines — Common Hardware Components

Turbomachinery	Combustion Devices
• Fuel Pump Housing Flow Paths	• Gas Generator Injector Interpropellant Plate
• Fuel Pump Impeller Flow Path	• Gas Generator Injector Housing
• Ball and Roller Bearings	• Gas Generator Combustion Chamber
• Turbine Outer Seals	• Gas Generator Combustion Chamber Liner
• Tiebolt Shaft and Disks (Modified Blade Attachments)	• Tubular Nozzle
• Internal Labyrinth Seals	• Nozzle Inlet Manifold
• Major Flange Seals	• Nozzle Discharge Manifold
• Bolts, Nuts, Studs, Washers, Pins	• Main Injector Interpropellant Plate
• 1st and 2nd-Stage Impeller Castings	• Main Injector Housing
• Uniform Cross Section Static Housing Seals	• Main Injector Faceplate
• Inducer Retaining Bolts	• Igniter Assembly — Main Injector
• Blade Retaining Rings, Tip Seals	• Igniter Assembly — Gas Generator Main
• Spacers, Bearing Sleeves, Wave Washers Made from Same Forging or Identical Hardware	Chamber to Injector Flange, Seals, Fasteners
Engine Controls	*Engine Assembly*
• Engine Controller	• Ducting 80% Small Lines 80% Large Lines
• Engine and Component Instrumentation	• GO_2 HEX
	• POGO Suppressor
	• Fuel Inlet Flex Joints
	• Fasteners, Seals

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The two-stage methane pump operates at 10673 rpm to deliver fuel at the required pressure of 4621 psia. From the pump exit the fuel flows through the fuel shutoff valve (FSOV) and to the chamber/nozzle cooling jacket manifold where the flow splits so that 25 percent goes to the regenerative nozzle cooling jacket and 75 percent goes to the regeneratively cooled chamber jacket. The nozzle cooling flow is used entirely to fuel the gas generator while the chamber coolant flow is discharged at 409 R directly into the main chamber injector.

The high-pressure oxidizer pump operates at 7601 rpm to provide the oxygen pressure level of 3338 psia required by the cycle. From the pump exit, approximately three percent of the LO_2 flow is diverted to the gas generator oxidizer control valve and subsequently to the gas generator. The bulk of the LO_2 flow (97 percent) flows through the main oxidizer control valve and directly to the main chamber injector.

The high-pressure, high-temperature (1688 psia/1800 R) gas from the gas generator provides the power to drive the high-pressure propellant pumps. The hot gas flow is initially expanded through the methane turbine and is subsequently routed to a second turbine which powers the oxidizer pump. The turbine exhaust gas is then diverted through the gaseous oxygen heat exchanger (for tank pressurization) and then discharged through a nozzle of area ratio 5.0 to produce thrust.

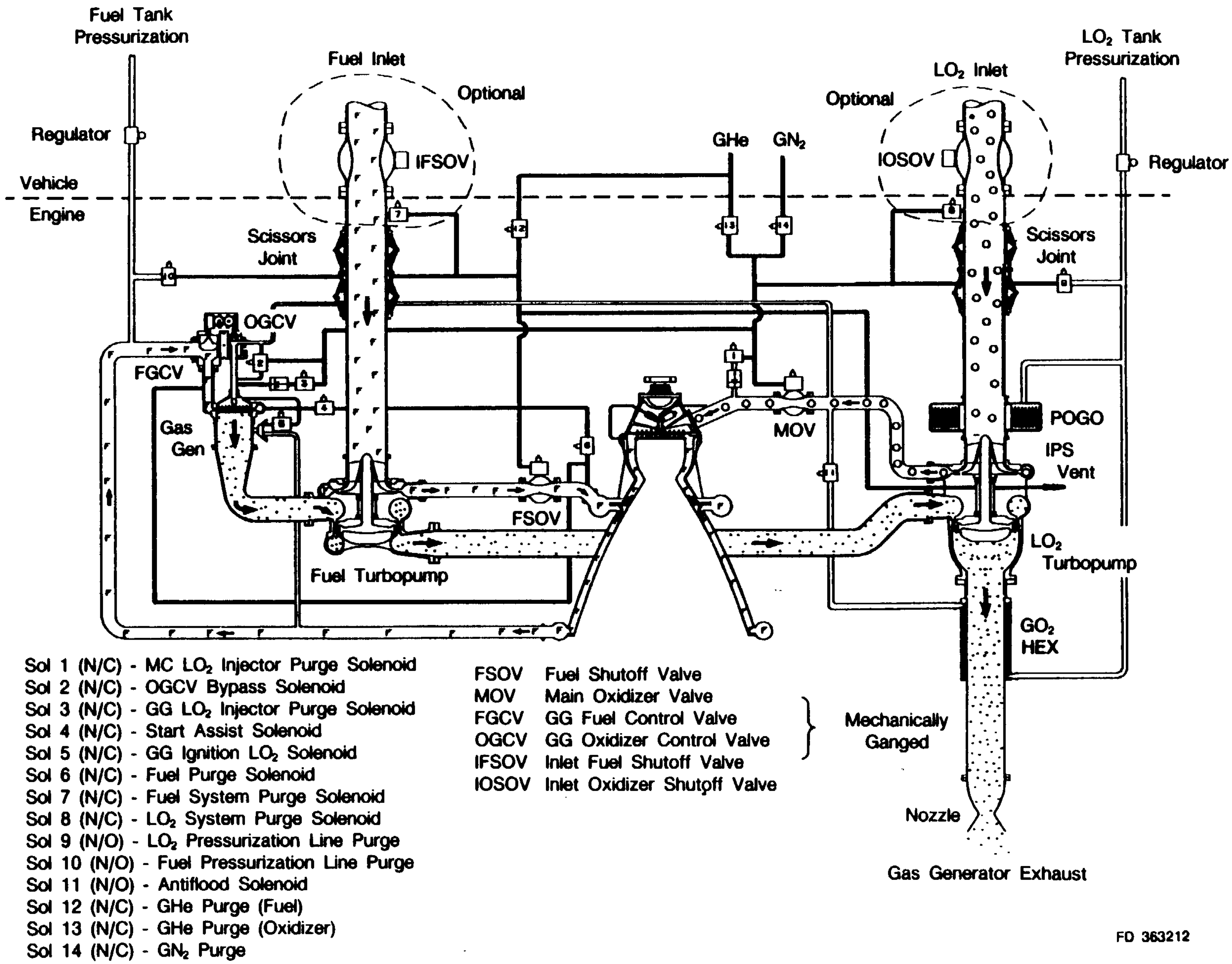

Figure 3.1.2-1. Simplified Flow Schematic for STBE Derivative Gas Generator Cycle Engine

3.1.2.2 Engine Operation

The engine will be preconditioned using liquid flow from the tanks to soak the turbopumps until they are sufficiently cooled. The inlet valves will be opened, allowing liquid from the tanks to flow down to the turbopumps and letting any vapors to percolate back up to the tanks to be vented.

The engine start is a timed sequence process using an oxidizer lead for reliable soft propellant ignition. The oxidizer lead avoids hazardous buildup of unburned fuel in the combustor or on the pad, because all fuel is consumed immediately upon injection. Reliability of ignition is enhanced by the LO_2 lead because the transient mixture ratio during propellant filling includes the full excursion of ignitable mixture ratios from greater than 200 to less than one.

With the oxidizer lead start sequence, the GG LO_2 injector is primed prior to opening the GG fuel valve to assure liquid oxidizer flow, thus eliminating turbine temperature spikes due to oxidizer phase change. After the GG LO_2 valve is opened, the main oxidizer valve (MOV) is opened followed by both the fuel GG valve and the fuel shutoff valve (FSOV). Helium spin assist is provided to the gas generator to help start the turbopump rotating and is discontinued early in the engine acceleration. Gas generator and main chamber ignition is accomplished with common design dual electrical spark-excited, oxygen/methane torch igniters. Engine acceleration is accomplished by open-loop scheduling of the gas generator oxidizer control valve.

Main stage thrust control is provided through open loop control of the GG oxidizer valve. Engine mixture ratio is preset by trim of the main oxidizer valve.

Engine shutdown is accomplished using a time based scheduling of the propellant valves. The gas generator oxidizer valve is closed first to power down the turbopumps, then the GG fuel valve is shut along with the MOV. The FSOV is closed when the pump is at low rpm. Provisions for post shutdown purging of propellants is provided.

3.1.3 Turbomachinery

3.1.3.1 Oxidizer Turbopump Hardware Description

The oxidizer turbopump is configured as a single-stage centrifugal shrouded impeller pump with an inlet inducer driven by a two-stage axial flow turbine. The design features of this turbopump are shown in Figure 3.1.3-1. The inducer and impeller, made of fine grained cast and Hot Isostatically Pressed (HIP) Inconel 718, is coupled to the turbine through a single turbine disk with an integral shaft made of Waspaloy. Pump and turbine inlet and discharge housings are fabricated from fine grained cast and HIP Inconel 718 to minimize machining costs. Turbine blades and vanes are made from cast Mar-M-247 nickel base alloy. The ball and roller bearings, made of 440C material will be used to support the pump rotor system. Investigations are ongoing to find an alternate cyrogenic bearing material or combination. Any data and/or technology that is obtained through this investigation or the Space Shuttle Main Engine Alternate Turbopump Development (SSME-ATD) program, will be applied to the STME and STBE turbopump bearings and bearing systems.

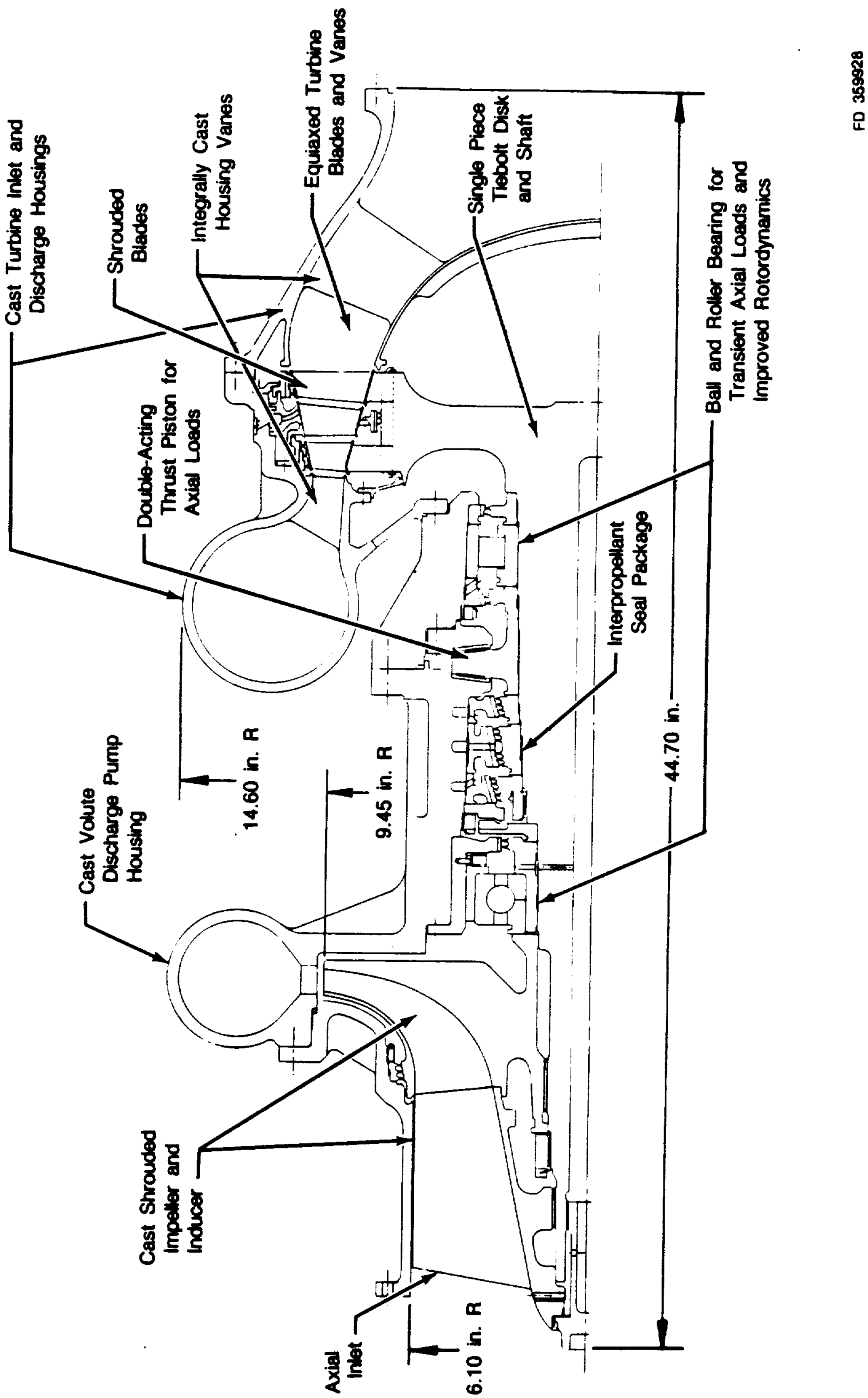

Figure 3.1.3-1. STBE Derivative Gas Generator LO_2 Turbopump

The rotor thrust balance system is accomplished through the incorporation of a double acting thrust balance system on the turbine side of the interpropellant seal turbopump in a liquid fuel environment so as to avoid any rub in a LO_2 environment. Externally supplied high pressure fuel (methane) is used for thrust piston actuation and for roller bearing and turbine coolant. The rotating thrust piston is made of forged Inconel 718 and its mating surface of the stationary housing is an insert made of Bearium B-10 material (leaded bronze). Axial travel of the rotor is controlled at this location.

The double-acting thrust piston provides thrust balance capability to the rotor system by controlling axial imbalance loads during startup, steady-state, and shutdown operation. As an axial imbalance load occurs, the rotor moves axially, which opens an orifice that supplies high-pressure fuel to the side of the piston in which the rotor has traveled. At the same time, the opposite piston face is now vented to low pressure fuel, resulting in a reaction thrust load that restores the rotor to its initial position.

While the roller bearing is cooled by fuel, the ball bearing is cooled with LO_2. Oxidizer flow along the backside of the impeller is used as bearing coolant, then is recirculated to the inducer inlet through a controlling orifice/hole in bearing carrier and shaft. Bearing DNs for the ball and roller bearings is 0.88×10^6 and 1.06×10^6 respectively. In addition, damper seals will be used to assist in rotor damping.

The interpropellant seal package employs a labyrinth seal design with a helium buffer cavity. This design is similar to the SSME ATD LO_2 turbopump design. An oxidizer-side vaporizer is incorporated to reduce the overboard leakage.

The turbine inlet housing is a cast volute integrating the first-stage turbine vane, and contains the placement of the turbine tip seal lands. A gas-cooled liner is not required at this location because of relatively low temperatures and pressures as compared to the fuel turbopump turbine inlet. Attachment of the inlet housing to the pump housing is achieved with a flexible arm designed to provide thermal compatibility between the two housings.

The turbine discharge housing is a fine grained cast and HIP Haynes 230 configuration which incorporates an exit guide vane. This guide vane is required, due to the relatively high second-blade exit angle, to avoid excessive flow losses resulting from redirecting the flow into an axial direction.

3.1.3.2 Fuel Turbopump Hardware Description

The fuel turbopump is configured as a two-stage centrifugal shrouded impeller pump with an inlet inducer driven by a two-stage axial flow turbine The design features of this turbopump are shown in Figure 3.1.3-2. The inducer and impeller, made of fine grain cast titanium A-110 ELI, are coupled to the turbine through an integral turbine disk shaft made of forged Waspaloy. Pump and turbine inlet and discharge housings are fabricated from fine grained cast and HIP Inconel 718 to minimize machining costs. Turbine blades and vanes are made from cast Mar-M-247 nickel base alloy. The ball and roller bearings, made of 440C material, will be used to support the turbopump rotor system. In addition, damper seals will be used to assist in rotor damping. These fluid hydrostatic bearings are supplied with leakage flows from the impeller back face.

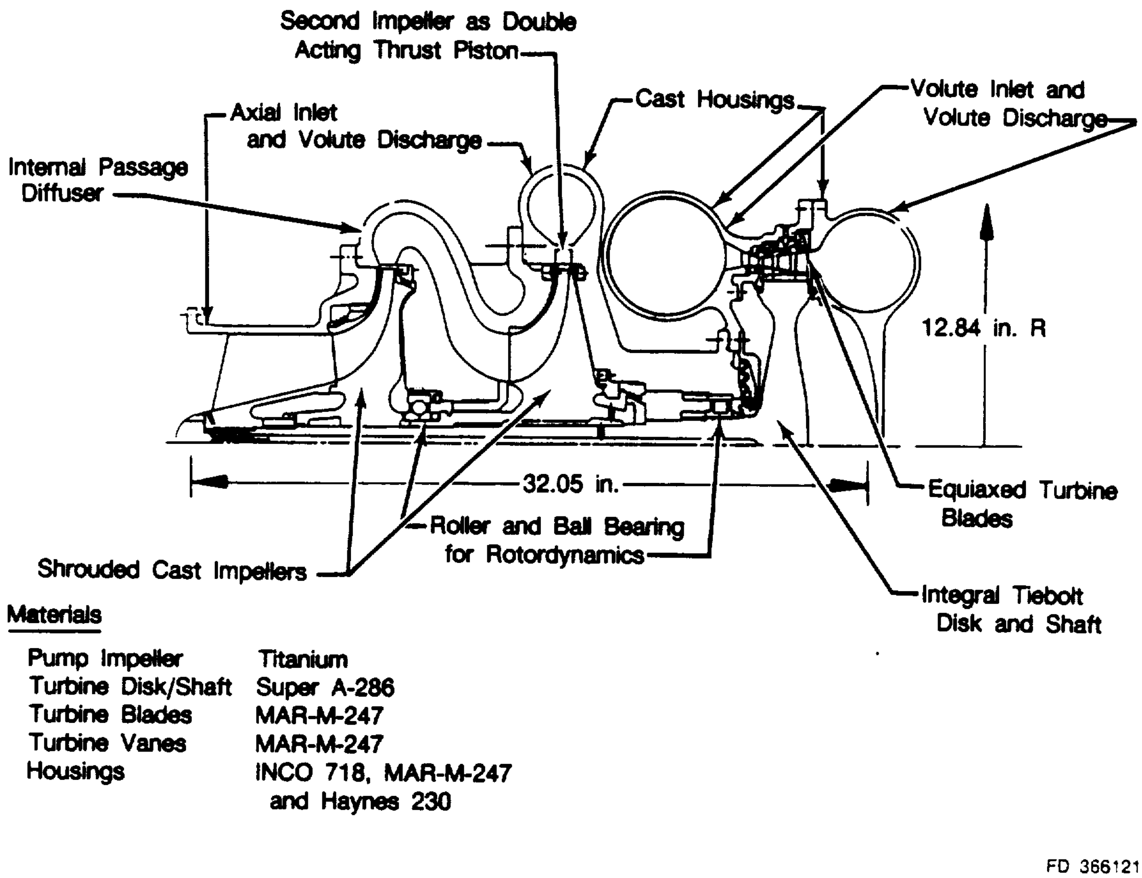

Figure 3.1.3-2. STBE Derivative Gas Generator Fuel Turbopump

The rotor thrust balance system is accomplished by incorporating the thrust piston into the second-stage impeller. A double acting, double orifice thrust piston has been configured into the front and back side of the impeller. The thrust piston is designed to control axial imbalance loads during engine startup, steady-state, and shutdown conditions. As the thrust imbalance load occurs, the rotor moves axially, which then opens an orifice at the impeller tip, introduces pump discharge high pressure fuel to the side of the impeller in which the rotor has traveled. At the same time, the opposite impeller face is vented to low pressure fuel, resulting in a reaction thrust load to restore the rotor axial position. Both sides of the thrust piston are fed with second-stage impeller discharge pressure. Axial travel is limited by a forward stop on the impeller ID shroud face and by an aft stop on the ID of the impeller back face.

The ball and roller bearings are the same bearings used on the SSME-ATD fuel turbopump. The ball bearing is cooled by first-stage discharge pressure bled off the impeller back face and flow controlled by the labyrinth seals near the outer diameter of the impeller. The roller bearing is cooled by second-stage discharge pressure that is supplied to the bearing via internal passages through the pump housing. Roller bearing coolant is then discharged into the turbine disk cavity to be used as turbine coolant. Bearing DNs for the ball and roller are 0.64×10^6 and 0.78×10^6 respectively.

The turbine inlet and discharge housings are fine grain and HIP casting Haynes 230 volutes. Attachment of the inlet housing to the pump housing is achieved with a thermally compatible designed flexible arm.

A diaphragm type lift-off seal (similar to the ATD fuel turbopump) is incorporated in the fuel pump design to prevent cooldown flow from entering the turbine during the pre-start sequence. At engine start, pump pressure increases so that lift-off seal is deflected and flow is permitted through the bearing and into the turbine for additional turbine cooling requirements.

3.1.3.3 Turbomachinery Rotordynamics

The P&W Advanced Launch Systems (ALS) Program is designed to produce reliable, low-cost rocket engine turbopumps. Pratt & Whitney uses proven design criteria and analytical methods in the design of rotordynamic operation for jet engine rotors and rocket engine turbopumps. Each Derivative Gas Generator Oxygen Turbopump (DGGOT) and Derivative Gas Generator Fuel Turbopump (DGGFT) design incorporates configuration changes which result in stiffer rotors, bearings, and rotor support structures with the addition of roughened stator damper seals. For optimum rotordynamics, each rotor is supported by strategically located, stiff, durable bearings. These changes result in a significant improvement to the first fundamental bending mode of the rotor, moving it well beyond the maximum operating design speed. This, in addition to an improved rotor balance procedure, results in an effective low speed balance of the rotor for low synchronous response. Rotor stability in the DGGOT and the DGGFT have been improved by designing the turbopumps to operate below the first vibrational mode of the rotor. Increased stability margin in each turbopump is provided by the introduction of roughened stator damper seals into the design.

A critical speed summary for the DGGOT is provided below.

W_{cr} (rpm)	% Design Speed	% Rotor Strain Energy	Mode Description
7016	99	7.3	Pump Pitch
14556	199	6.4	Turbine Bounce
54067	740	95.5	1st Bending

A critical speed summary for DGGFT is provided below.

W_{cr} (rpm)	% Design Speed	% Rotor Strain Energy	Mode Description
11829	110	22.2	Pump Bounce-Turbine Pitch
16509	154	9.8	Pump Pitch-Turbine Bounce
26671	249	82.0	1st Bending

3.1.4 Combustor

Injector Elements

The injector element performance is critical to the combustion efficiency and stability of the combustion process. Two important parameters relating to the injector performance and design are pressure drop and the number density of elements on the injector face.

The ΔP across the elements must be high enough to prevent "chug" or fuel system coupled instability (minimum six percent P_c for fuel and four percent P_c for LO_2). The ΔP is also important to the drop size distribution produced by the element and hence the combustion efficiency of the chamber. The element density sets the overall dimensions of the coaxial injection elements which must stay within manufacturing and contamination limits. In addition to these considerations, the derived STBE engine needed to be designed using the same injection elements as used in the STME unique chamber design. The actual design set for the gas generator cycle STME and derived STBE injector is given in Table 3.1.4-1. This injector meets the above design constraints in both engines. The injector is estimated to produce a LO_2 droplet spray with a 55 micron MMD in the STBE engine based on coaxial injector performance data recently taken for the National Aerospace Plane program by Pratt & Whitney. The main penalty involved in using the same injector elements in both the STME and derivative STBE is that a higher pressure drop is required in the STBE than would otherwise be required for injector element designs for that engine alone.

Table 3.1.4-1. *STME/Derivative STBE Injector Elements Dimensions and Operating Conditions*

	Derivative STBE	*STME*
Chamber Pressure-psi	2250	2250
Fuel Flow-lb/sec	442.3	164.7
ΔP Fuel-psi	293.0	170.0
LO_2 Flow-lb/sec	1541.0	1112.0
ΔP LO_2 psi	302.0	157.0
No. of Elements	890	890
Spud ID-in.	0.272	0.272
Annular Gap-in.	0.02	0.02

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Acoustic Liner

The derivative STBE combustor chamber will be provided with an acoustic liner to suppress combustion instability. The liner consists of a perforated surface that absorbs a portion of a reflected pressure wave, thereby damping the intensity of the reflected wave and decoupling the wave from the combustion process. A common measure of liner performance is the absorption coefficient which is equal to the energy absorbed divided by the incident wave energy. The absorption for a given liner design and operating conditions can be calculated by the P&W acoustic liner design deck.

The acoustic liner design proposed for the derivative STBE core is listed in Table 3.1.4-2. To arrive at this design some of the parameters, such as the acoustic aperture hole diameter and length, had to be estimated. These parameters are usually set by the cooling channel dimensions and have a relatively small impact on acoustic absorption. The 0.05 area ratio (acoustic hole area/total acoustic liner area) was set based on past parametric studies which have shown this value to be close to optimum. The backing cavity depth was set to maximize absorption at the first tangential frequency of the combustion chamber (1423 Hz). Experience has shown that this is the most likely frequency of combustion instability. The liner placement in the chamber (near the injector face) and length are based on experimental testing and design experience which has shown that combustion can be stabilized by ¼-length liners with a minimum of 20 percent acoustic absorption at the frequencies of interest. Further experimental verification of the acoustic liner design procedures and assumptions such as backing cavity gas temperature will be obtained in the planned testing of the STBE subscale test chamber under contract NAS8-37490. The predicted acoustic absorption of the STBE acoustic liner as a function of frequency is shown in Figure 3.1.4-1. The curve shows good acoustic absorption over a broad frequency range.

Table 3.1.4-2. STBE Derivative Engine Acoustic Liner Design and Operating Conditions

Chamber Pressure-psi	2250.0
Aperture — Gas Temperature-°R	2000
Aperture — Gas Molecular Wt.	22.2
Hole Diameter-in.	0.1
Hole Length-in.	0.35
Area Ratio	0.05
Backing Cavity Depth-in.	0.60
Liner Length-in.	4.8

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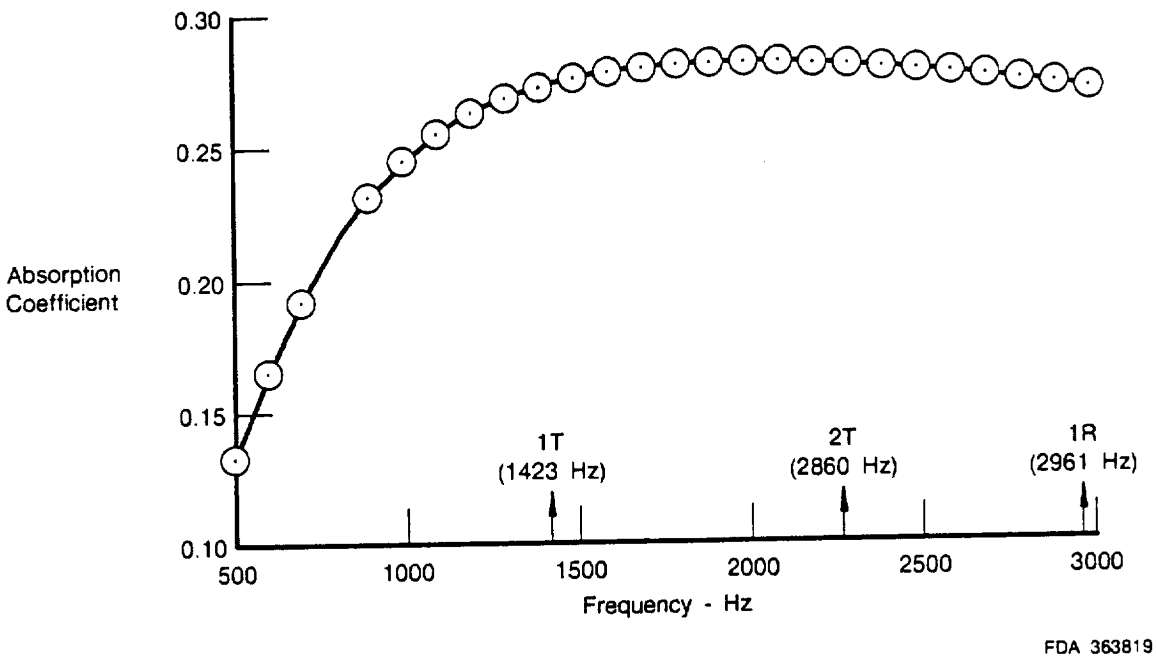


Figure 3.1.4-1. Sound Absorption vs Frequency for STBE Derivative Gas Generator Acoustic Liner

3.1.4.1 Main Injector

The main injector design uses 869 coaxial, tangential entry injection elements arranged in a hexagonal concentric pattern in a 20.35 inch-diameter injector face. This type of injector element has consistently demonstrated efficient, stable combustion in all of the P&W high-pressure combustion programs. The main injector assembly is shown in Figure 3.1.4-2.

The oxidizer injection element, shown in Figure 3.1.4-3, is a tube which is closed at one end and has a 0.272-inch ID and a 0.020-inch wall thickness. The oxidizer is introduced into the tube through three slots that are oriented on a tangent to the tube ID. The tangential entry produces a hollow cone spray of liquid oxygen which results in extremely fine atomization, and rapid, stable combustion.

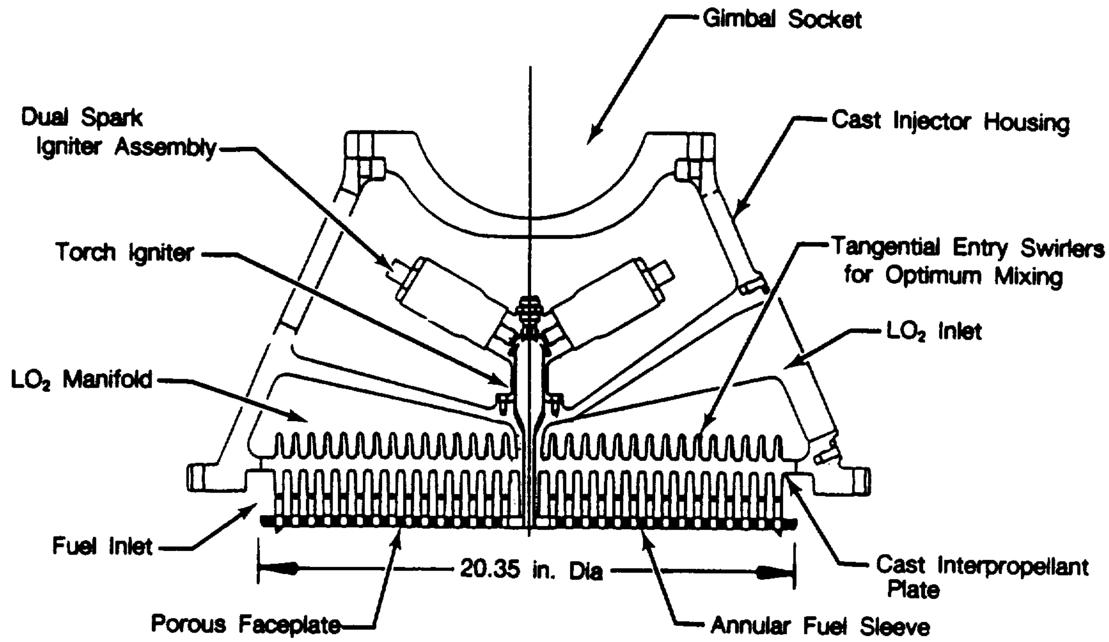


Figure 3.1.4-2. STBE Derivative Gas Generator Main Injector Assembly

Fuel is introduced through an annulus surrounding each LO_2 injection element. The annulus is formed by the fuel sleeve which is cast integral with the injection element and brazed to the porous faceplate. Fuel enters the injector from the combustion chamber coolant interface, and flows radially inward in the injector manifold which is formed by the interpropellant divider plate and the porous faceplate. At each LO_2 injection element, the fuel is directed into the individual fuel annuli by four radial slots in the fuel sleeve. The fuel is then discharged from a 0.02 $in.^2$ annulus surrounding each LO_2 injection element. The faceplate is fabricated from a porous material, woven wire product consisting of Haynes 230 cobalt alloy, allowing approximately five percent of the fuel which is introduced into the injector to flow through the injector face to achieve faceplate durability.

The main injector assembly is fabricated from fine grained cast and HIP Inconel 718 with cast injection elements integral with the propellant divider plate. The injector design provides for a center-mounted torch igniter and also is configured to contain the engine gimbal thrust structure.

3.1.4.2 Combustion Chamber

The combustion chamber is regeneratively cooled by fuel from the high-pressure pump discharge. The fuel enters the thermal skin cooling jacket at the regeneratively cooled nozzle manifold chamber interface. The coolant then flows forward, counter to the gas path flow, to the throat. The fuel cools the chamber wall, exits at the injector interface internal manifold, and enters the injector. This flow configuration provides the coolest fuel at the throat where wall heat flux is highest. The combustion chamber is shown in Figure 3.1.4-4.

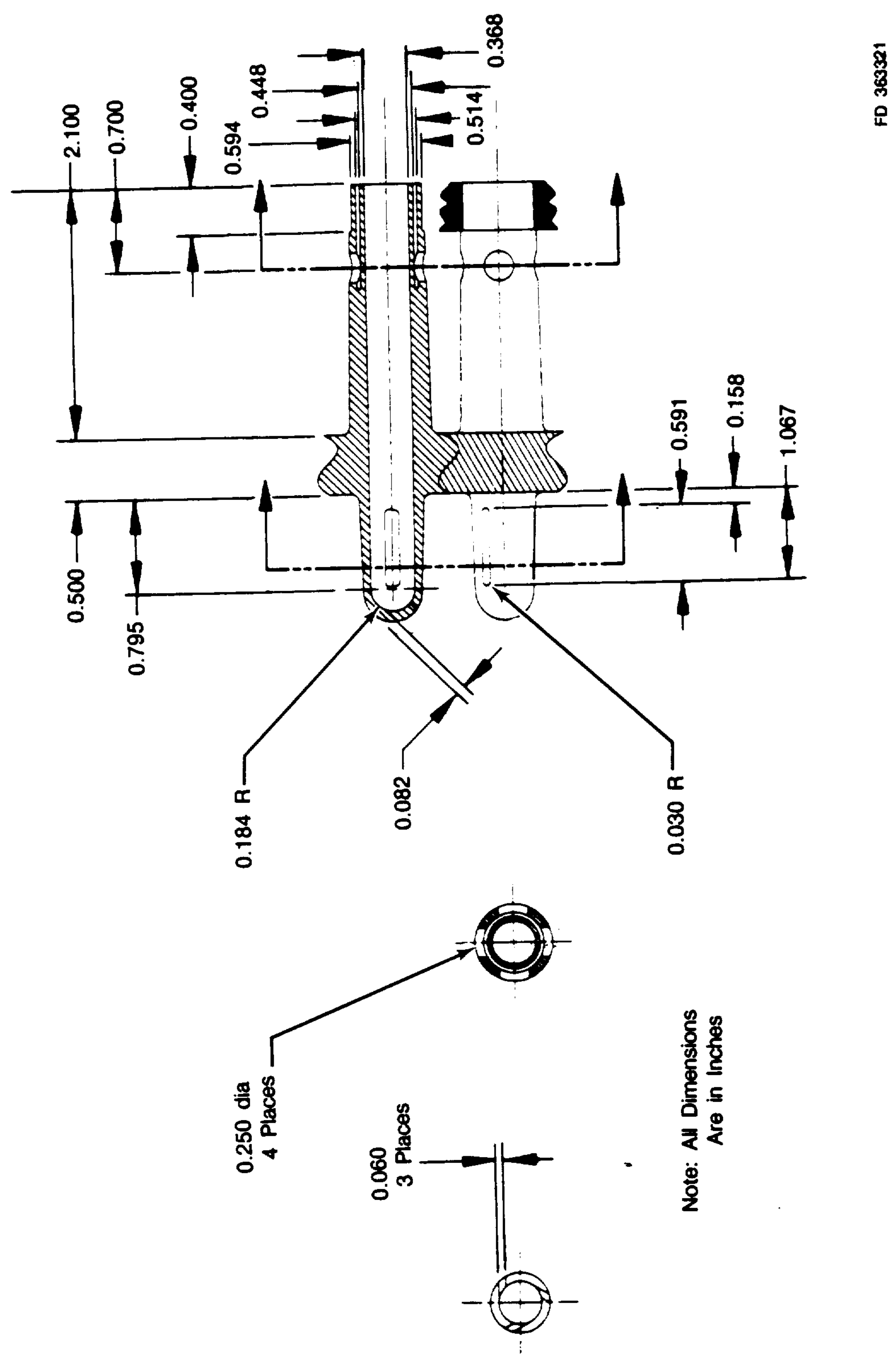


Figure 3.1.4-3. STBE Derivative Gas Generator Main Injector Element

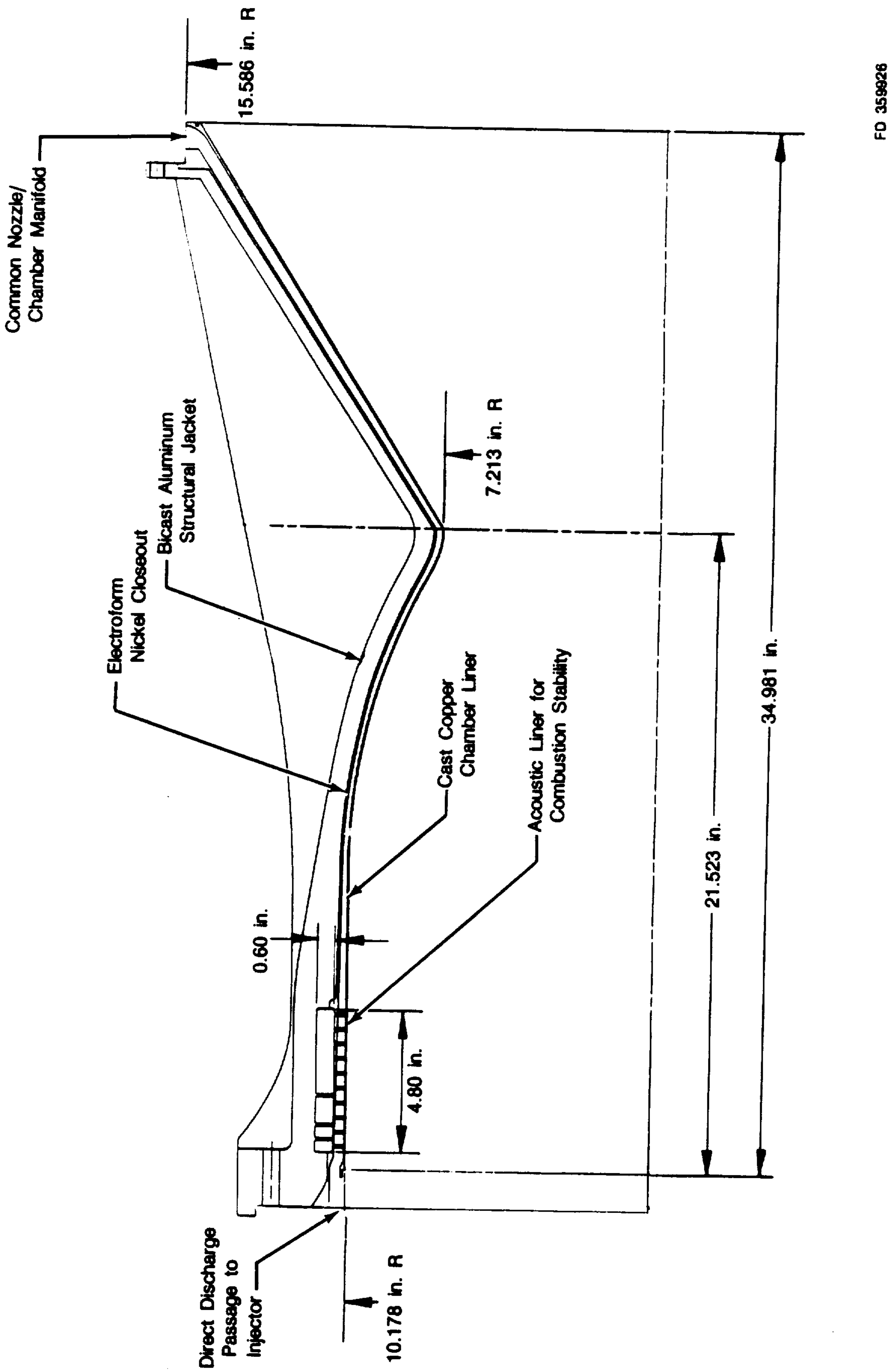


Figure 3.1.4-4. STBE Derivative Gas Generator Combustion Chamber

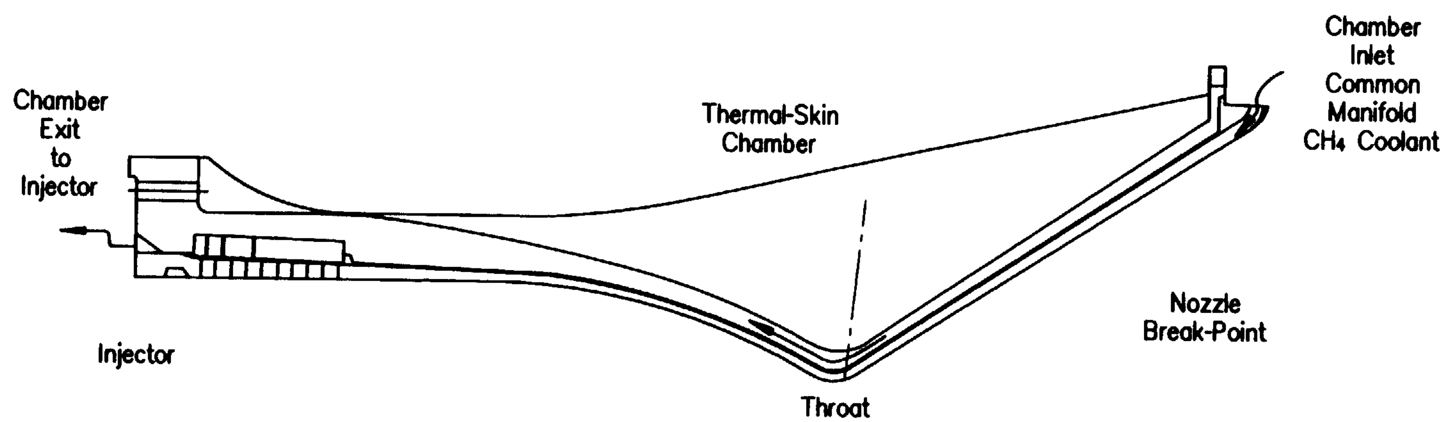
The main combustion chamber uses similar construction technologies as the SSME Main Combustion Chamber in the area of the regeneratively cooled liner. However, the construction differs in the structural jacket design. The regeneratively cooled liner will be forged from NASA-Z copper alloy. The cooling passages are machined from the copper alloy liner and an electrodeposited nickel close-out is applied which forms the outer jacket of the liner. At this point the structural jacket of aluminum is installed around the liner by a bi-cast method. This is accomplished by positioning a sand mold around the liner, then the structural jacket is cast in place with an aluminum casting alloy.

The structural aspect of the bi-cast chamber design is very similar to the conventional welded nickel design. The layer of copper and nickel is used to close out the passages and hold the coolant pressure, and the structural jacket is used to contain the hoop loads due to the combustion chamber pressure. The axial load from the nozzle, i.e., thrust, is also transferred through the jacket by longitudinal webs in the bi-cast aluminum design. A close fit between the copper liner and structural jacket is obtained to ensure that the hoop loads are transmitted to the jacket and do not cause overstressing of the liner.

An acoustic cavity is positioned adjacent to the injector face to provide combustion stability. The acoustic cavity is located behind the copper alloy liner. The cavity is connected to the combustion chamber cavity through a specified number and size of holes through the liner between the coolant passages. A liner is placed in the acoustic cavity to which a minimal amount of coolant flow is tapped off the chamber coolant exit, and used to cool the backside of the acoustic cavity. This coolant is then dumped into the cavity to provide a purged outflow, preventing hot gas ingestion into the acoustic cavity.

The STBE derivative gas generator thrust chamber is a derivative of the STME chamber. The derivative chamber has identical values of manifold location and size, divergent nozzle exit diameter, chamber diameter and injector-to-nozzle exit length as the STME design. The chamber features a machined passage thermal-skin NASA-Z liner/nickel closeout assembly surrounded by a structural jacket. The coolant enters the common inlet manifold and flows counterflow toward the injector, where it discharges directly into the injector. The chamber inlet manifold is common with the tubular nozzle which improves the inlet geometry and reduces inlet pressure drop. Since the chamber is cooled with all the chamber fuel flow, the exit manifold can be eliminated which minimizes the coolant exit pressure drop. Due to the higher thrust requirement of the STBE (636K lbf), the throat diameter has been increased from the STME value of 12.87 inches to 14.43 inches. The chamber contraction ratio of 2.0 is less than that for the STME as a result of maintaining a common injector diameter while increasing the throat diameter. The inclusion of the acoustic liner in the chamber increases the difficulty of cooling the liner with this reduced contraction ratio. To cool the liner within the cycle requirements, the number of passages has been set at 330 with a maximum passage height-to-width aspect ratio of 5.0. The cooling at the throat has been further improved by designing for coolant side curvature enhancement of the heat transfer film coefficient. Figure 3.1.4-5 presents the derivative thrust chamber contour and passage geometry summary.

The coolant passage dimensions have been sized to meet the heat transfer and cycle requirements at the 120 percent thrust design point of 750K thrust and the chamber pressure of 2250 psia. Figure 3.1.4-6 summarizes the predicted thrust chamber cooling performance at the 120 percent thrust design point. The chamber liner has been designed so that the maximum hot wall temperature is approximately 1530 R. The maximum wall heat flux at this wall temperature is 55.2 Btu/in.^{2}-sec which occurs one-inch forward of the throat. The coolant side curvature enhancement at the high heat flux point is approximately 35 percent. The coolant enters the liner at 236 R and 4934 psia and exits at 430 R and 2589 psia. The passage geometry has also been sized so that the coolant never exceeds a Mach number of 0.5. The highest Mach number in the derivative chamber is 0.2.



Chamber Contour Data

Chamber Length = 21.523 in.
Divergent Nozzle Length = 13.46 in.
Throat Diameter = 14.43 in.
Injector Diameter = 20.34 in.
Contraction Ratio = 2.0
Divergent Nozzle Area Ratio = 2.16
L^* = 41.0 in.
ηc^* (Throat) = 0.98
Number of Passages = 330
Liner Construction - Thermal-Skin
Liner Material - NASA Z

Cooling Passage Geometry

Axial Length (in.)	Wall Radius (in.)	Passage Width (in.)	Passage Height (in.)	Land Width (in.)	Wall Thickness (in.)
-21.5	10.18	0.083	0.375	0.111	0.035
-18.1	10.18	0.083	0.375	0.111	0.035
-16.1	11.18	0.083	0.335	0.111	0.035
-12.1	11.18	0.120	0.205	0.074	0.035
-9.1	10.06	0.120	0.186	0.071	0.035
-5.0	9.17	0.102	0.195	0.090	0.035
-1.0	7.39	0.085	0.173	0.054	0.035
-0.0	7.21	0.085	0.168	0.052	0.035
2.4	8.70	0.085	0.340	0.081	0.047
5.4	10.58	0.140	0.324	0.073	0.061
10.4	15.71	0.140	0.467	0.121	0.086
13.5	15.59	0.140	0.480	0.157	0.100

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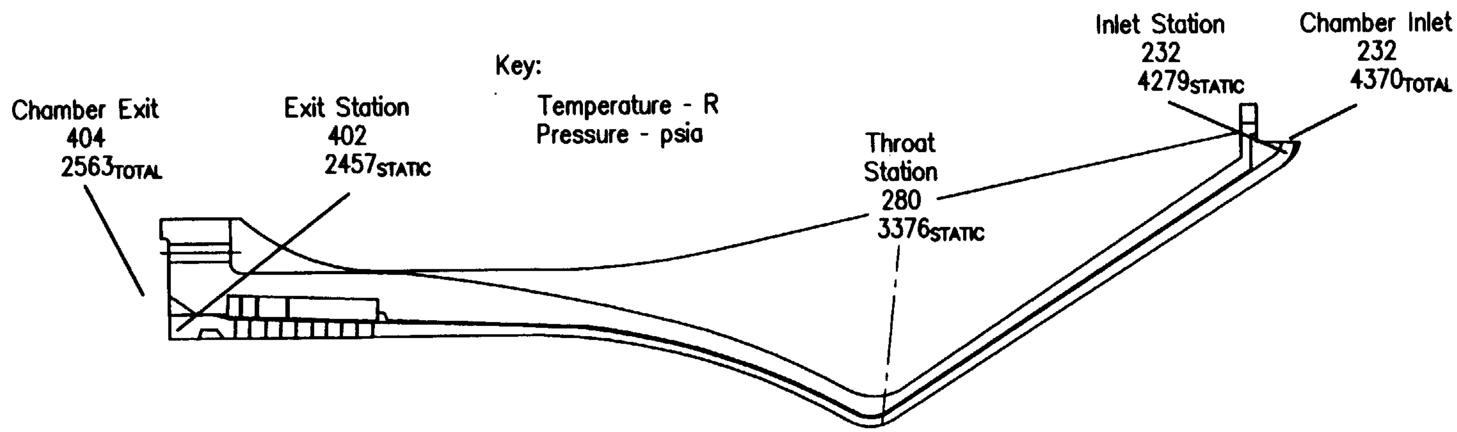
Figure 3.1.4-5. STBE Derivative Gas Generator Chamber Cooling Passage Geometry and Contour Data



-21.5	-18.1	-16.1	-12.1	-9.1	-5.0	-1.0	0.0	2.4	5.4	10.4	13.5
932	1471	1506	1496	1435	1411	1502	1502	1289	1509	1446	1377
16.4	30.3	32.5	34.2	35.6	41.1	54.6	55.2	23.3	17.7	12.9	11.2

Hot Wall Temperature & Heat Flux

Key:

Axial Location - in.
Wall Temperature - R
Heat Flux - Btu/in.^{2} - sec

Coolant Performance

Thrust = 120%
M_{cool} = 442.2 lbm/ sec

Chamber Heat Transfer Performance

Thrust - lbf	637K
Chamber Pressure - psia	2250
Coolant Flow - lbm/sec	442.2
Inlet Temperature - R	232.0
Exit Temperature - R	407.0
Coolant Heat Pickup - Btu/sec	67653.0
Inlet Pressure - psia	4370.0
Exit Presure - psia	2563.0
Pressure Drop - psid	1807.0

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Figure 3.1.4-6. STBE Derivative Gas Generator Chamber Heat Transfer Performance Summary

3.1.4.3 Torch Igniter

A continuous burning torch igniter was chosen for use in both the gas generators and main combustion system because of the simplicity of the design and reliability in tests. The igniter configuration employed evolved from development efforts since 1957 at Pratt & Whitney and is based on experience gained from the successful RL10 and XLR-129 engine programs.

In the gas generator, the torch is mounted in the combustor wall, two inches axially from the injector face, and expels the hot torch combustion gases at a right angle to the flow path from the gas generator injector, thus providing safe, efficient, reliable ignition of the combustion system. In the main combustion chamber, the torch is mounted axially in the center of the injector, directing the torch down along the centerline of the combustion chamber.

3.1.4.4 Gas Generator Combustion System

The gas generator employs a fixed-area injector which injects the fuel and liquid oxygen to provide hot gas for the turbopump turbines. This injector design is the result of experience in hot firings using three generations of high-pressure 250K preburner injectors. Approximately 96 percent of the fuel is injected through the concentric annuli around each oxidizer element. The remaining fuel passes through a porous faceplate to provide transpiration cooling and to hold the combustion process off the faceplate. The gas generator assembly is shown in Figure 3.1.4-7.

Liquid oxygen is supplied to the injector from the gas generator oxidizer valve to the injector manifold/dome. Oxidizer flow is injected into the combustion chamber through 199 individual swirler elements. Each element has flow entries machined tangentially to the inner diameter. The fuel is injected into the combustion chamber through radial slots in the element fuel sleeve.

The gas generator injector is fabricated from a cast Haynes 230 divider plate with integral injection elements. The oxidizer manifold cavity is formed by a bolted-on dome-shaped end plate. The fuel manifold is formed by a torus welded to the cast divider plate. The porous faceplate is brazed to each injector element fuel annulus sleeve, thereby providing structural support to the plate. The faceplate is made from a porous, woven wire product consisting of Haynes 230 cobalt alloy. This material provides good oxidation resistance and high temperature strength to resist the erosion effects if hot gas scrubbing does occur. The faceplate provides a high enough pressure differential to cause the fuel to uniformly distribute for concentric injection into the sleeve around the oxidizer element, yet passes enough fuel to transpiration cool the material and float the combustion gas away from its surface.

The combustion chamber consists of two basic assemblies, the scrub liner and the structural case. The scrub liner forms the hot gas flowpath and protects the structural case from the hot gas. The scrub liner consists of a porous and non-porous liner. Both are made from Haynes 230 cobalt base material needed for its oxidation resistance and high-temperature capabilities. The front three inches consists of a porous liner that is transpiration cooled by a portion of the fuel flow tapped from the gas generator injector. This front zone is the region of highest energy release, and in addition to providing thermal protection, the porous liner also serves as an effective acoustic damper to prevent combustion instability. The other (non-porous) liner is a cylindrical duct which forms the combustion chamber.

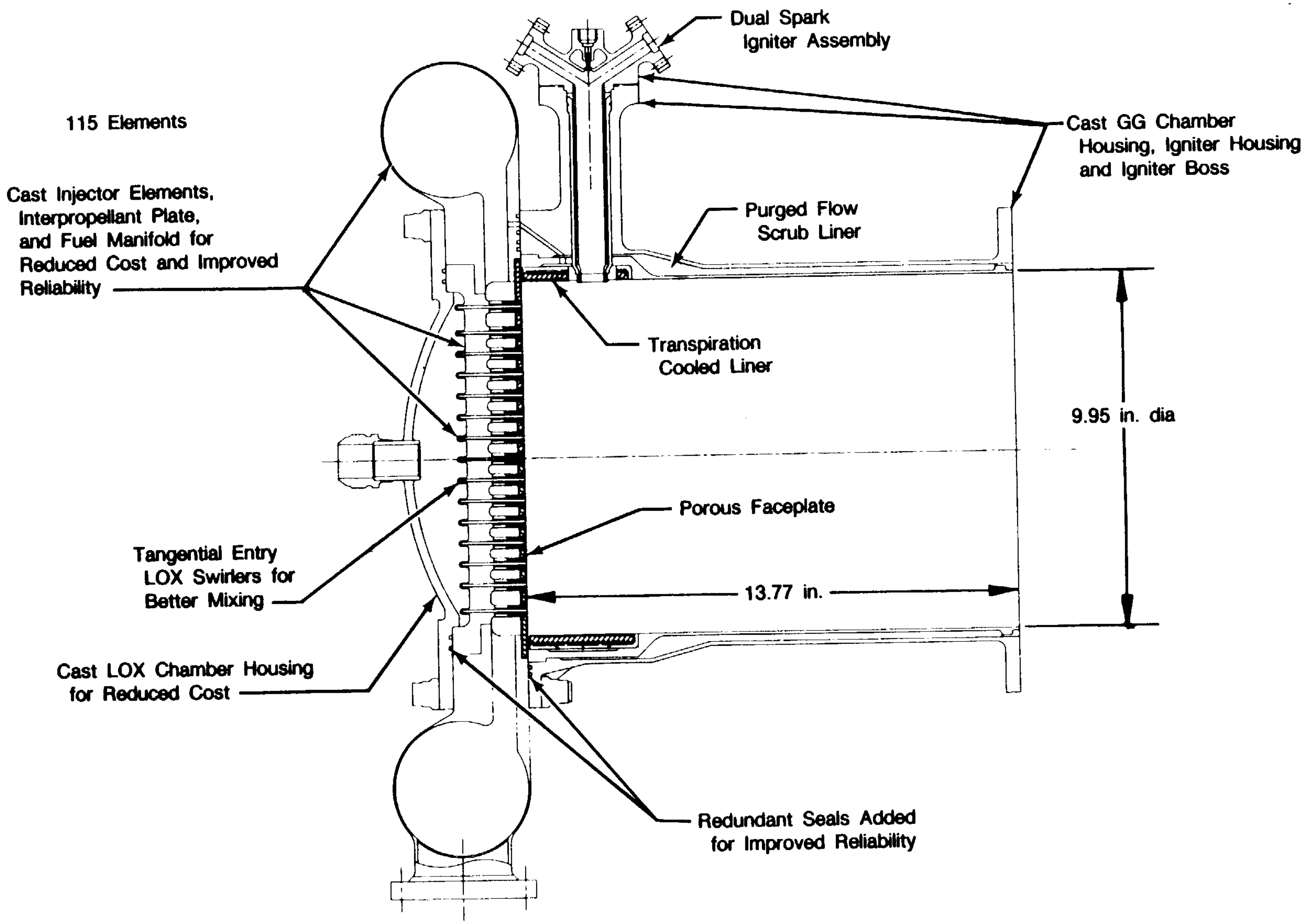

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Figure 3.1.4-7. STBE Derivative Gas Generator Assembly

3.1.5 Nozzle

3.1.5.1 Regeneratively Cooled Nozzle

The regeneratively cooled nozzle, shown in Figure 3.1.5-1, is constructed from 990 SPIF (Super Plastic Inflation Formed) tubes of AISI 347 stainless steel, surrounded by a structure shell of closed cell elastomeric foam with a filament wound composite overwrap. This shell is also designed to carry all hoop loads.

The SPIF nozzle is welded to the inlet and exit manifolds which are both made of AISI 347 SST. The closed cell polyurethane foam on the exterior of the nozzle, would adhere to the nozzle surface and act as a compliant layer between the nozzle and the composite structural shell due to the coefficient of expansion difference between the nozzle and shell. At cryogenic operation the foam would become rigid, thereby transferring the nozzle hoop load into the structural shell. The nozzle coolant inlet manifold supplies coolant to the nozzle and the combustion chamber, making the nozzle parallel coolant flow and the combustion chamber coolant counterflow.

The regeneratively cooled nozzle is entirely common with that of the STME. Figure 3.1.5-2 summarizes the nozzle geometry. The nozzle is constructed of 990 super plastic inflation formed AISI 347 stainless tubes. The nozzle is 56-inches long and extends from an expansion area ratio of 2.16:1 to an exit area ratio of 27.9:1. The number of passages and the passage diameters have been sized so that the operating stresses of the wall never exceed the 0.2 percent yield stress. An alternate nozzle design could be constructed of 990 Haynes 230 tubes.

The coolant enters the nozzle at an area ratio of 2.16, flows parallel to the gas path flow and exits at an area ratio of 27.8. Figure 3.1.5-3 presents the predicted heat transfer performance of the nozzle. the nozzle is cooled with 146 lbm/sec of fuel that enters at 234 R and 4024 psia and exits at 563 R and 3502 psia. The maximum hot wall temperature and heat flux are 1455 R and 10.9 Btu/in.^{2}-sec, respectively.

3.1.6 Gas Generator Engine Control

The STBE control consists of sensors, interconnects, a controller, actuators, propellant valves, ancillary valves, and a health monitor. The functional layout of the STBE control components is shown on Figure 3.1.6-1. The controller time sequences the valves for engine control and maintains engine safety by sensing hazards and taking corrective action. A single electromechanical actuator drives both the gas generator fuel and oxidizer valves. The main chamber oxidizer and fuel shutoff valves are helium actuated. The gas generator fuel and oxidizer valves use similar sleeve valves, and the main chamber oxidizer and fuel shutoff valve use similar poppett valves. The health monitor is integrated with the controller but electrically isolated to prevent health monitor faults from propagating into the controller and jeopardizing engine safety.

Engine thrust is regulated by trimming the gas generator oxidizer valve while engine mixture ratio is regulated by trimming the main oxidizer valve. Oxidizer flow shut-off is provided by the gas generator oxidizer valve and the main oxidizer valve while positive fuel flow shut-off is provided by the main fuel shutoff valve.

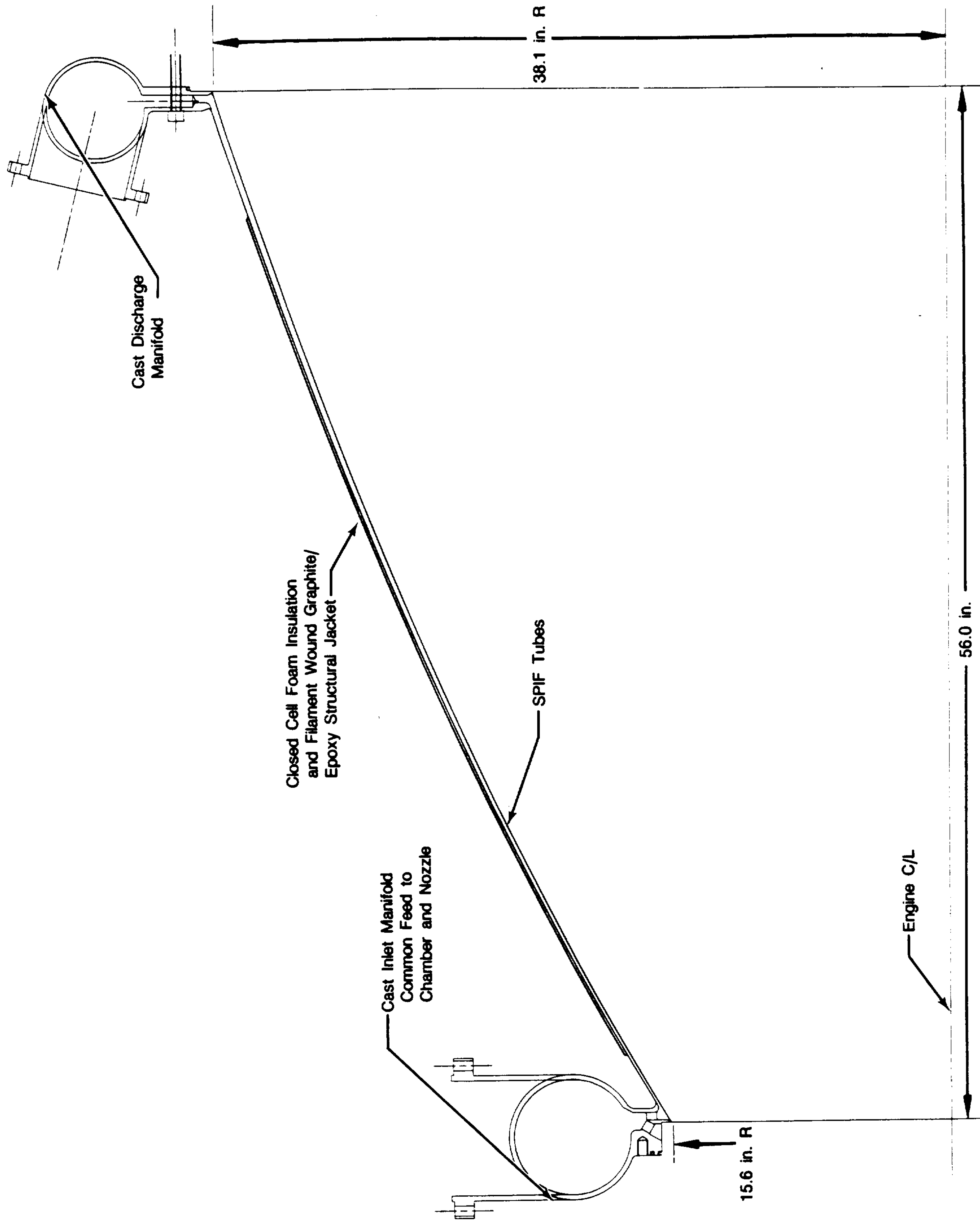
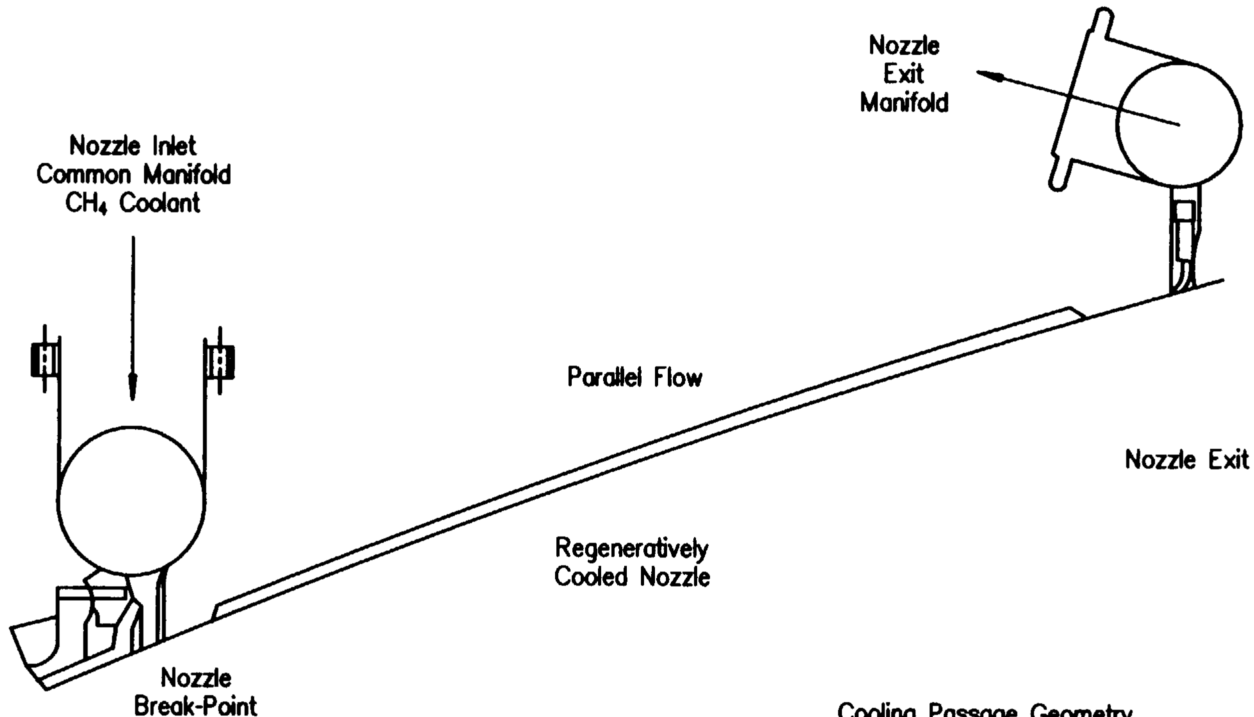


Figure 3.1.5-1. STBE Derivative Gas Generator Regeneratively Cooled Nozzle



Nozzle Contour Data

Nozzle Length - 56 in.
Inlet Expansion Ratio - 4.65
Exit Expansion Ratio - 27.87
Number of Passages - 990
Nozzle Construction - Super Plastic Inflation Formed
Nozzle Material - AISI 347

Cooling Passage Geometry

Axial Length (in.)	Wall Radius (in.)	OD Width (in.)	OD Height (in.)	Wall Thickness (in.)
13.5	15.59	0.097	0.105	0.02
25.9	22.11	0.139	0.152	0.02
40.3	28.36	0.179	0.196	0.02
54.7	33.51	0.211	0.232	0.02
69.5	38.08	0.241	0.266	0.02

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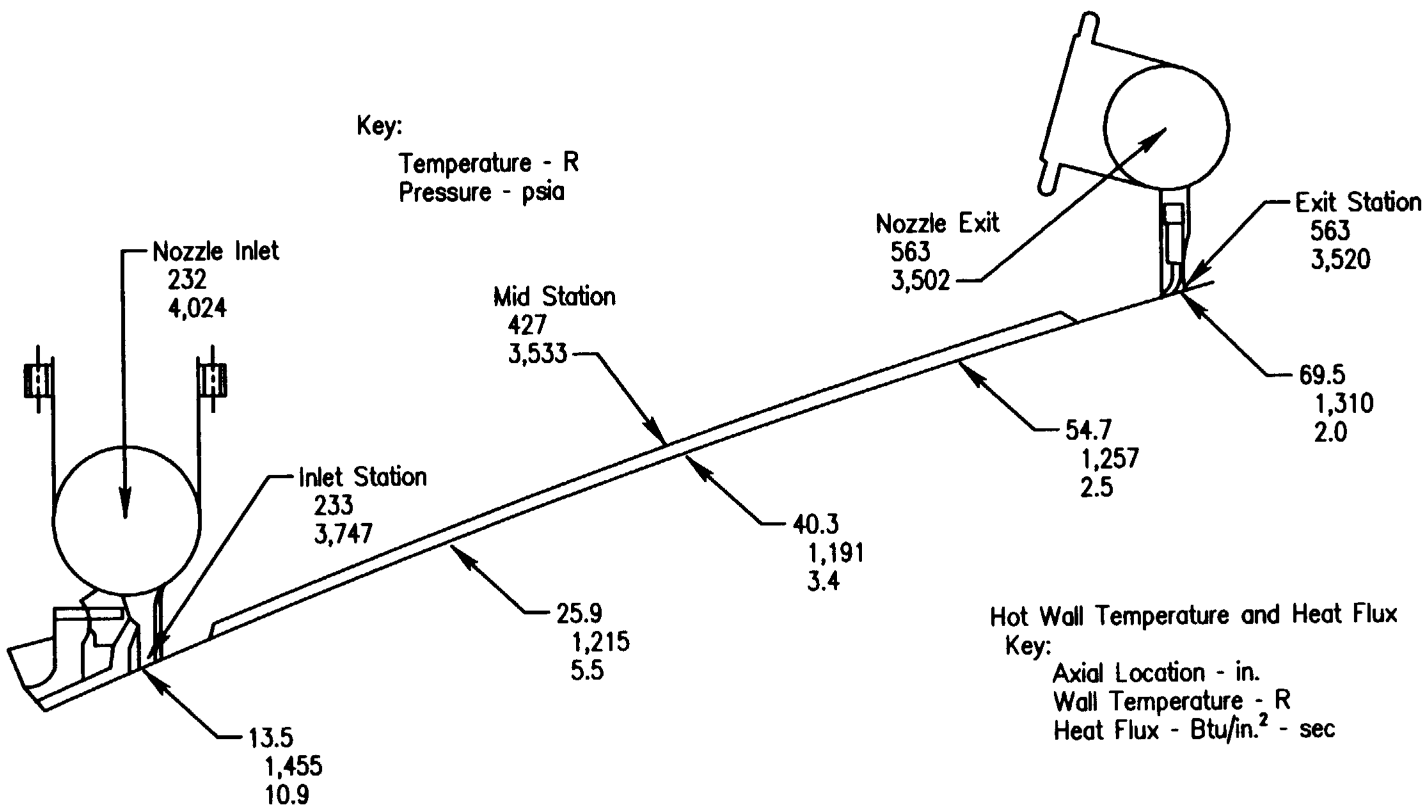
Figure 3.1.5-2. STBE Derivative Gas Generator Nozzle Cooling Configuration



Coolant Performance
- Thrust = 120%
- M_{cool} = 146.0 lbm/sec

Nozzle Heat Transfer Performance

Thrust - lbf	645K
Chamber Pressure - psia	2,250
Coolant Flow - lbm/sec	146.0
Inlet Temperature - R	234.0
Exit Temperature - R	563.0
Coolant Heat Pickup - Btu/sec	41,683.0
Inlet Pressure - psia	4,024.0
Exit Pressure - psia	3,502.0
Pressure Drop - psid	522.0

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Figure 3.1.5-3. STBE Derivative Gas Generator Nozzle Heat Transfer Performance Summary

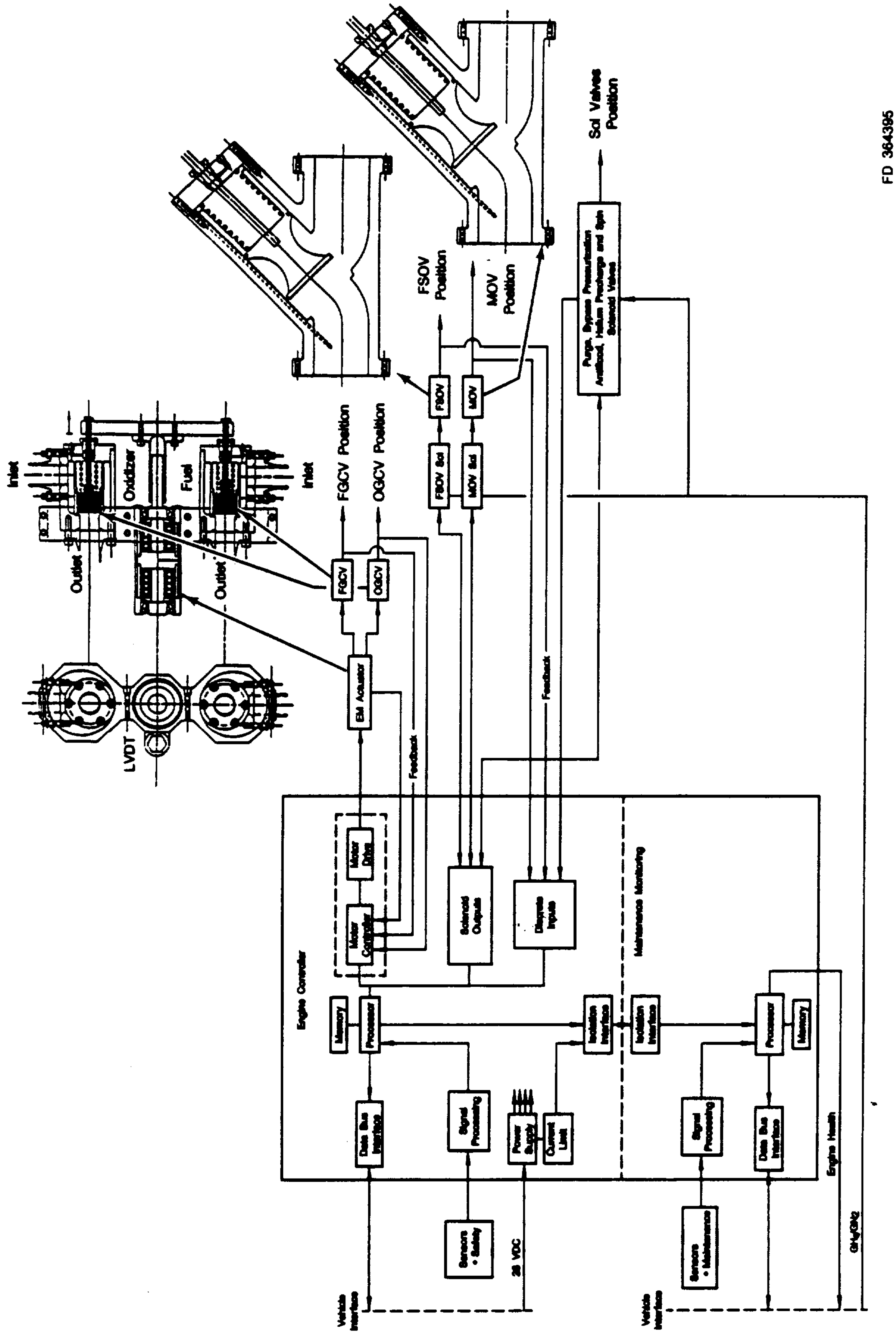

Figure 3.1.6-1. STBE Derivative Gas Generator Engine Control and Health Monitor System Functional Concept Meets All Requirements With Low-Cost Approach

Requirements used to establish a control and monitoring system concept are shown in Table 3.1.6-1.

Table 3.1.6-1. *Control System Requirements*

Requirement	*Engine Requirement*	*Control System Requirement*
$300/lb Launch Cost	Low Recurring Costs	Design for Low Costs and Reliability, Provide Prelaunch Checkout
Design Life	5 Hours, 30 Missions	Durability, Maintenance Monitoring
Reliability	Demonstrate 0.99 at 90% Confidence	0.9992
Safety	Fail Safe	Benign Shutdown
Thrust (Vac)	712K $\pm 3\%$	Ground Trim
Mixture Ratio	6 $\pm 3\%$ at 712K	Ground Trim
Transients	Start to 712K < 5 sec Max Rate of Change of Thrust Shutdown Impulse	Response TBD TBD
Interfaces		
• Tank Pressurization	GCH_4, GO_2	Valves, Logic
• Information	TBD	Data Bus, Baud Rate
• Electrical	N/A	28 vdc
• Ancillary Fluids		
–Ground Operation	Cooldown, Purge	He
–Vehicle Operation	Purge, Actuation	He

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3.1.6.1 Control/Health Monitor Conceptual Architecture

Conceptually the controller/health monitor is comprised of two functions: (1) control and safety monitoring and 2) maintenance monitoring. Control functions are those required to start, maintain normal operating conditions and shutdown the engine. Safety monitoring consists of real time engine evaluation to determine if an emergency shutdown is required. Maintenance monitoring looks at functional and physical characteristics which include many that are not flight critical, but real time definition is necessary to properly schedule maintenance.

The STBE engine uses a simplex, full authority digital electronic engine control with dual channel input/output (I/O). A single channel control with an effector system designed to direct engine shutdown upon loss of controller function meets the fail safe design requirement. Controller reliability requirements are met with dual I/O interfaces which receive inputs from dual sensors with the information being processed by a single microprocessor.

The output interface supports solenoids with dual windings and a dual channel electromechanical actuator interface. One of the two solenoid windings in each device has the capacity for solenoid operation in the event that one winding fails opens. Shorted solenoid switches are accommodated by switching both high and low sides of the solenoid. The electromechanical actuator (EMA) interface is a dual active effector system with single processor control. Under normal conditions, each output interface provides one half the drive signal necessary for actuator control. If one of the EMA interfaces becomes inoperative, the current drivers in the inoperative interface are depowered and the gain in the remaining interface is doubled to provide full control capability. This dual active interface provides smooth transfers from dual channel operation to single channel operation.

Actuator loop failure detection is provided by current wraparound, feedback failure detection, and open-loop detection. Current wraparound is provided by measuring actuator winding current and comparing the result to the requested value.

Feedback failures occur if the actuator position sensors produce an erroneous result to the controller. Feedback failure detection is provided by detecting out-of-range readings or detecting a difference between the dual sensor readings. Open-loop detection is provided by comparing the requested actuator position to the measured position. The error between the request and feedback is measured over a period of time and compared to a threshold value. If the measured actuator error is above the threshold value, an open-loop failure is declared. In the event that an actuator malfunction cannot be isolated to a given interface, an engine shutdown is effected by the logic.

An initiated built-in-test (IBIT) mode is provided by the controller to detect faults during prestart. In the IBIT mode, the controller sequences solenoid valves and electromechanical actuators throughout their operating range. This feature enhances mission reliability by providing a low cost method for testing the system prior to launch.

The health monitoring system works as an interface between the electronic control, engine sensors, and the vehicle avionics while transmitting real time data to the Vehicle Health Monitoring System (VHMS). Safety monitoring is performed by the electronic control with any performance or anomaly information passed to the maintenance monitoring unit through an isolation interface. Instrumentation not critical to flight operation is processed by maintenance monitoring electronics. Maintenance monitoring information is transmitted to the vehicle independently of the control.

3.1.6.2 Controller Hardware Approach

Highlights of the control/health monitoring system architecture include modular design of the engine control functional requirements. The system level design includes control of discrete inputs and outputs (solenoids and switches), actuator positioning, sensor signal processing and control law processing. This system design is implemented using state of the art hardware which provides a low risk, low cost flexible control.

Current plans are to provide a control design that meets reliability requirements with Class B components. By using these MIL-STD components and proper redundancy management, the reliability requirements can be achieved without the cost penalty of Class S components. With the advent of microelectronics, multiple channel controls are viable options without paying a significant weight penalty. Multiple channel controls will be considered during Phase B as a way to improve life cycle cost.

3.1.6.3 Vehicle Interface Definition

Independent vehicle interfaces are supported by both the engine control and health monitor. Independence is necessary to ensure faults in the maintenance data bus from causing a fault in the control data bus. These data buses will be designed to be compatible with the vehicle data bus selection. The only identified differences will be those that address flight criticality. The engine controller interface will be updated to meet different flight safety requirements.

Isolated interfaces between control and maintenance monitor were selected to support the integrated design concept. The key to these interfaces is to incorporate failure containment regions. Failure containment is accomplished through design.

3.1.6.4 Actuators/Valves

An extensive trade study was conducted to select valve and actuator types based upon an assessment of cost, reliability, performance and hardware commonality. Low cost was ranked as the primary selection criteria with manufacturability, design simplicity and maintainability all being considered cost drivers. The study considered pneumatic, hydraulic and electromechanical actuators as well as sleeve, poppett, ball, and butterfly valves. From this study, the following configurations were selected.

3.1.6.4.1 Ganged Gas Generator Valves/Actuation

The ganged gas generator valve system consists of two valves and an electromechanical actuator. Oxygen and fuel flow to the gas generator are controlled by the Oxidizer Gas Generator Control Valve (OGCV) and Fuel Gas Generator Control Valve (FGCV), respectively. The valves have been ganged together to eliminate potential turbine overtemperature events caused by the OGCV allowing oxidizer flow into the injector following fuel flow shutoff by the FGCV. A linear electromechanical actuator sequences the fuel and oxidizer valves to acheive proper engine start, throttling, and shutdown. Additionally, an oxidizer gas generator bypass valve supplies five percent of oxidizer gas generator flow necessary for starting. This valve is separate from the ganged valve assembly and uses the same concept as the ancillary valves.

3.1.6.4.2 Oxidizer Gas Generator Control Valve (OGCV)

Operation

The OGCV is a modulating control valve that is located downstream of the oxidizer pump and upstream of the gas generator injector. Its function is to accurately control oxidizer flow into the gas generator and thereby control the thrust level of the engine. The valve schedules shown in Figures 3.1.6-2 and -3 indicate that the valve must accurately meter oxidizer flow for engine start, for engine transition to a second thrust level, and for engine shutdown, and therefore requires a high turndown ratio, or capability to meter accurately over a large range in flow. Evaluation of a valve type to meet these requirements at the lowest cost resulted in the selection of a right angle inlet to outlet translating sleeve type valve for this application, as shown in Figure 3.1.6-1. By contouring the sleeve metering ports, the valve area versus stroke relationship may be customized to meet the 2.5 percent accuracy requirement at all engine conditions. To meet the failsafe safety requirements for benign engine shutdown and to minimize required actuator force, the OGCV is pressure balanced and spring loaded in the closed direction. Thus, upon loss of actuator input force, for whatever reason, the OGCV slews to the closed position at a rate controlled by the valve force balance and the flow rate of oxidizer into the pressure balance cavity of the valve assembly.

Fabrication

The sleeve type OGCV design can be fabricated from standard bar stock shapes, allowing the use of simple manufacturing processes and ease of fabrication over a wide supplier base. Also, all parts/assemblies can be made identical to the FGCV with the exception of the sleeve, allowing low cost manufacturing due to increased lot sizes.

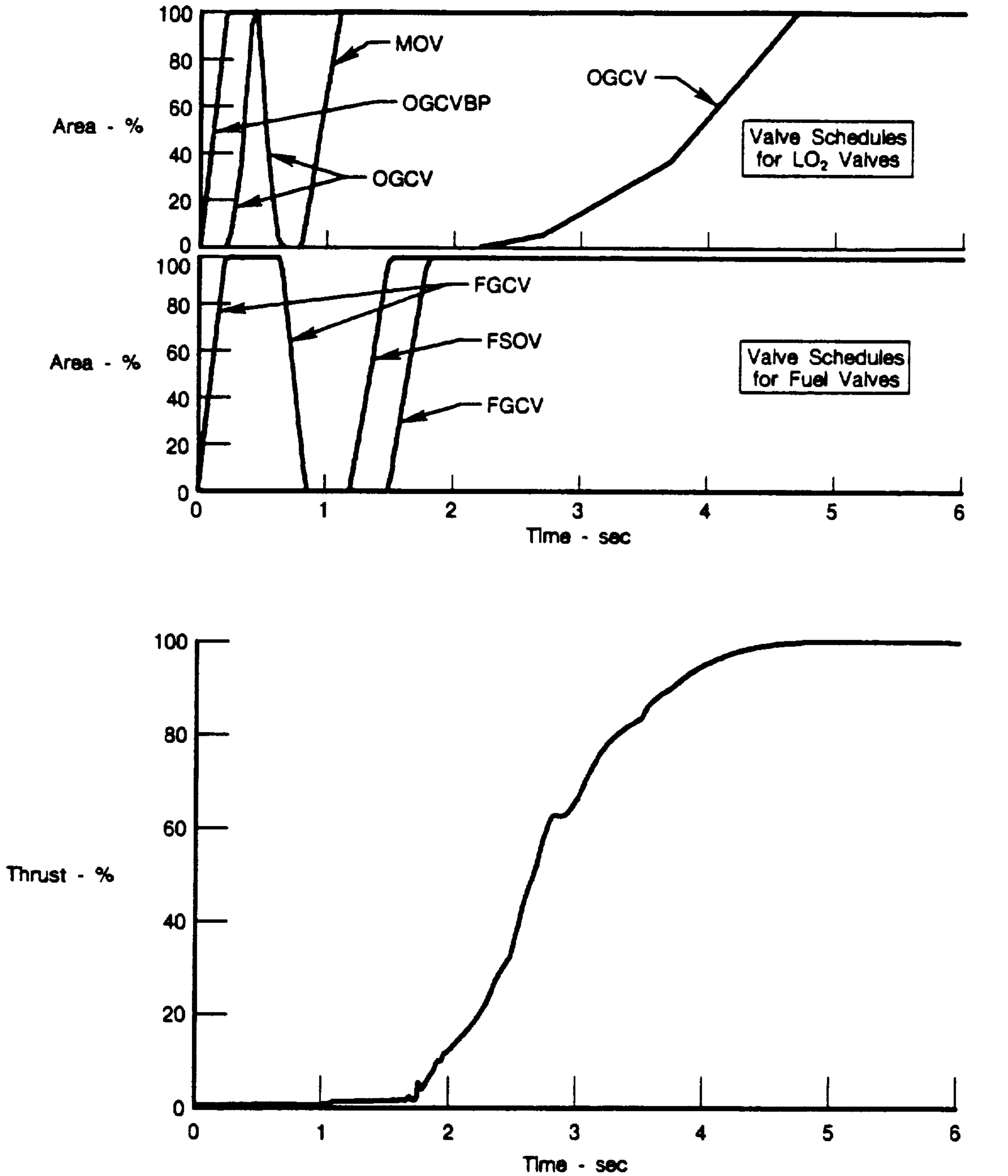

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Figure 3.1.6-2. Valve Sequence and Thrust Buildup for Engine Start

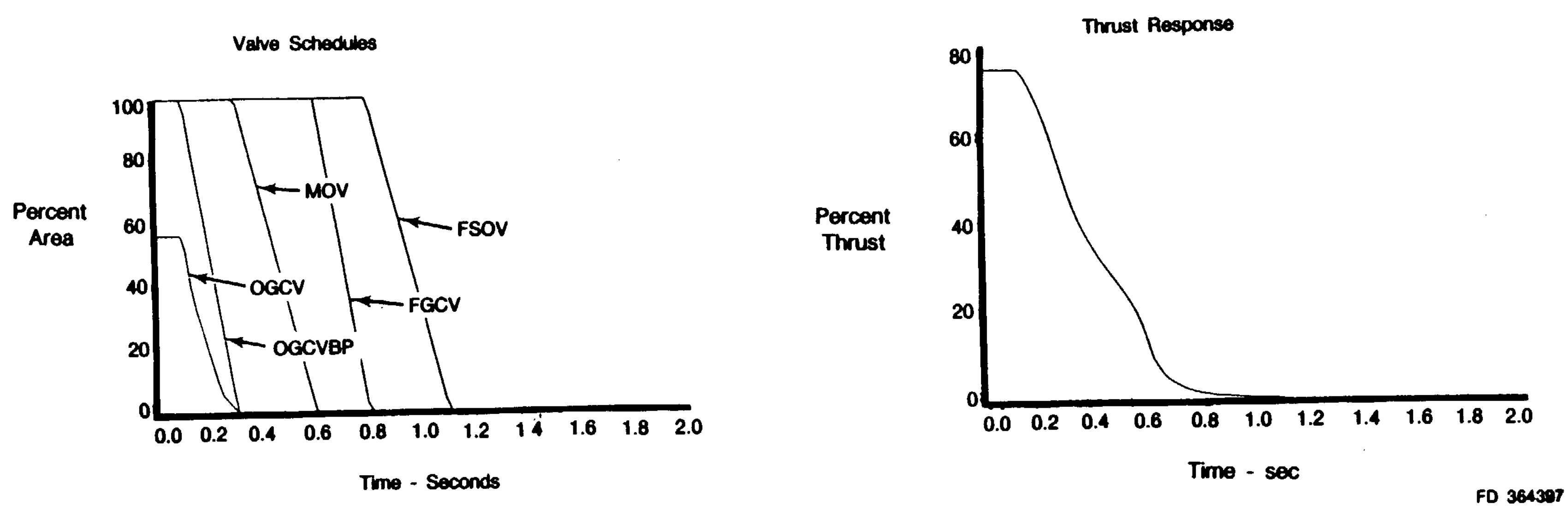

Figure 3.1.6-3. Valve Schedule and Thrust Transient for Engine Shutdown

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To reduce maintenance and improve reliability, ceramic materials are being investigated for the valve and sleeve elements. The material, Zirconia Toughened Alumina (ZTA), has been fabricated into a sleeve and valve configuration by a valve supplier. This valve eliminates the potential risk associated with metal to metal sliding surfaces in LO_2 and initial testing has shown that ZTA erosion and wear characteristics are ten times better than conventional 440 steel. Further investigation, including thermal shock testing, must be completed to determine this material's applicability.

3.1.6.4.3 Fuel Gas Generator Control Valve (FGCV)

The FGCV is an on/off valve that is located downstream of the nozzle fuel coolant exit and upstream of the gas generator injector. Its function is to control the flow of gaseous fuel into the gas generator and thereby control the gas generator oxidizer/fuel mixture ratio. To meet the engine start and throttling requirements the valve requires only one full open and one full closed position. Evaluation of a valve type to meet the requirements and provide maximum commonality with the OGCV has resulted in selection of a sleeve type valve identical to the OGCV with the exception of the sleeve which is ported for much higher area versus stroke gain. Since flow area is maximized when the sleeve ports are completely uncovered, the valve element may continue to translate without increasing the actual flow area of the FGCV. Thus, the ganged valve assembly may be postitioned variably to control OGCV position, which controls thrust, without impacting FGCV area. The FGCV is pressure balanced closed and spring loaded closed in a manner identical to that of the OGCV.

Fabrication

The FGCV will be fabricated identically to the OGCV and will reduce production cost by allowing larger lot size purchases of the identical FGCV and OGCV parts.

3.1.6.4.4 Ganged Valve Actuation

The gas generator valves are ganged for actuation with one actuator to eliminate potential turbine overtemperature events caused by the OGCV remaining open after the FGCV has closed. To satisfy the OGCV variable duty cycle this actuator must provide accurate position scheduling, while also providing a simple preflight checkout procedure. To meet the duty cycle requirements for both oxidizer and fuel flow, the ganged gas generator valves have been sequenced to result in actual area versus stroke characterisitcs as shown in Figure 3.1.6-4. This sequencing is permitted by the flexibility of the sleeve contouring and results in flow control as requested in the duty cycle. To provide a benign engine shutdown for the failsafe safety feature, the actuator must fail-passive such that the gas generator valve loading may backdrive the actuator to close both the OGCV and the FGCV. The lowest life cycle cost type of actuation which meets these requirements is electromechanical actuation. Since hydraulic fluid has been eliminated from the actuator, the operational cost of performing preflight checkouts is reduced and the cost of removal and replacement maintenance actions will also be reduced.

3.1.6.4.5 Electromechanical Actuator Operation

The electromechanical actuation system consists of a dual channel actuator controller and a linear ballscrew actuator. Electrical power is conditioned by a power conditioner to reduce the magnitude of the DC bus electrical transients and to prevent power surges from affecting module operation. The motor controller receives the position command signal from the engine controller along with the position signal from the actuator feedback module. The microprocessor-based controller provides signals to the motor drive circuit, consisting of appropriately configured power semiconductor switches such as metal oxide semiconductor field effect transistors (MOSFETs).

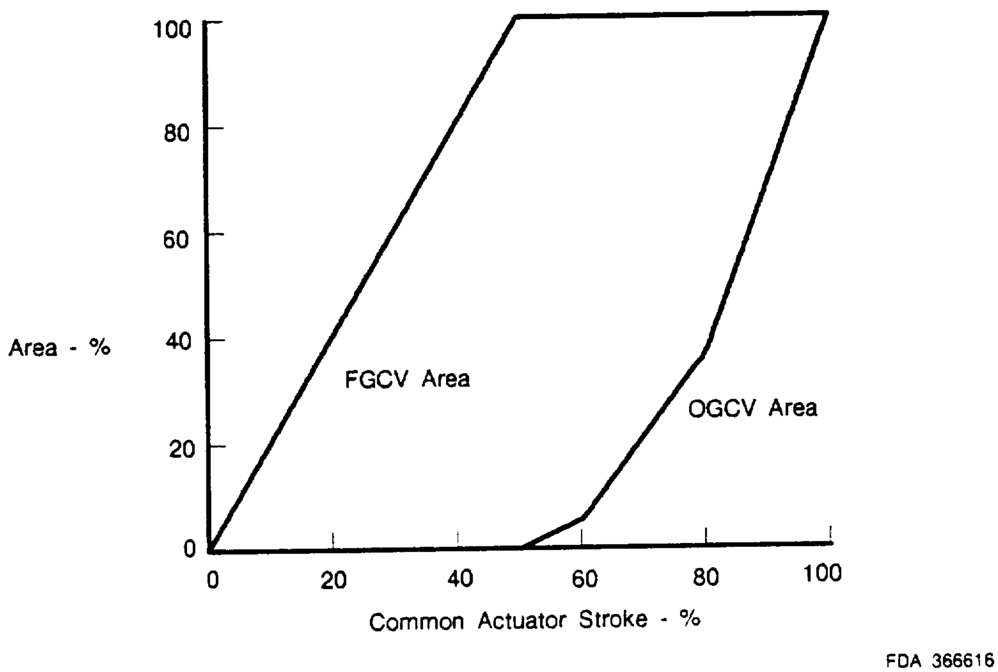

Figure 3.1.6-4. Schedule Requirements Feasible With Ganged Valves

The actuator module consists of dual switched reluctance motors (SRM) directly coupled to a ballscrew device. By directly driving the ballscrew with the electric motors the gear reduction element associated with electromechanical actuators may be eliminated. The electromechanical actuator linked with the gas generator valves is shown in Figure 3.1.6-1.

3.1.6.4.6 Main Oxidizer Valve (MOV)

The MOV is an on/off valve that is located downstream of the oxidizer pump and upstream of the thrust chamber. Its function is to control liquid oxidizer flow to the thrust chamber and thereby control the engine oxidizer/fuel mixture ratio. To meet the engine start and throttling requirements, the valve requires only one full open area position and a fully closed position. The valve must provide ±10 percent trimmability at the open position for engine mixture ratio trimming during the engine acceptance testing. A poppett valve has been selected as the lowest cost valve type which will meet all requirements. As shown in Figure 3.1.6-1, the poppett lends itself to precision trimming at the 90 percent open position, allowing accurate mixture ratio trimming. Since the valve has only two operating positions, full open and full closed, a translating helium piston actuator has been selected as the lowest cost option meeting all requirements. The actuator position will be controlled through a solenoid valve which is electrically scheduled by the engine controller. Discrete actuator position switches provide valve position feedback to the controller for preflight checkout as well as for in-flight operation.

MOV Option No. 1

To further reduce system cost and improve the reliability by removing components from the system, an optional propellant actuated MOV has been identified. The poppett valve may be pressure balanced and spring loaded such that a difference between the oxidizer pump inlet pressure and the pump outlet pressure serves as the actuation force on the MOV. This configuration restricts the MOV from easily being checked out during the preflight inspections, however, it reduces the potential of an uncommanded valve closure during main stage operation by removing the solenoid actuator and replacing it with a force balanced poppett assembly. Thus, the MOV will not close until the oxidizer pump pressure delta falls below 300 psid, eliminating the solenoid and actuator failure mode in which the pump is overpressurized as a result of MOV closure at main stage operation.

MOV Option No. 2

The MOV may also be electromechanically actuated to provide active mixture ratio trim during engine operation. Using the pressure balance technique, the valve loads may be reduced such that the electromechanical actuator used for the ganged valve assembly may also be used for the MOV.

3.1.6.4.7 Fuel Shutoff Valve (FSOV)

The FSOV is an on/off valve that is located downstream of the fuel pump and upstream of the nozzle and chamber coolant jackets. Its function is to control the total fuel flow into the engine cycle. To provide maximum cost benefit, a poppett type valve identical to the MOV has been selected. While pressure drop and weight could be improved using a ball valve design in this location, these factors have been traded for the simpler, lower cost poppett which also provides commonality with the MOV and the cost benefits which accompany commonality in development, production and logistics. The actuator is identical to that of the MOV providing additional system commonality. The actuator position will be controlled through a solenoid valve which is electrically scheduled by the engine controller. Discrete actuator position switches provide valve position feedback to the controller for preflight checkout as well as for in-flight operation.

3.1.6.4.8 Ancillary Valves

To provide propellant purging upon engine shutdown, tank pressurization during engine operation, pump interstage dam pressurization, and oxidizer gas generator valve bypass, solenoid actuated ancillary valves will be used. In each case the valves are low cost poppett type valves which require only short stroke actuation. For the propellant purge valves, a check valve is located between the poppett and the propellant line to help insure that the propellant is isolated from the helium system. These valves will incorporate commonality when possible, however, sizing and failsafe requirements for each valve must be defined before the degree of commonality can be established. Each ancillary valve will provide valve position feedback to the controller using dual valve open and valve close switches.

3.1.6.4.9 Operation

Valve/solenoid/ignition sequencing during prestart, start, mainstage, shutdown and post shutdown (in-flight) are shown in Figure 3.1.6-5.

3.1.6.4.10 Prelaunch Checkout

All valves are stroked from full closed to full open to full closed. Valve slew times provide verification that the valves are operational.

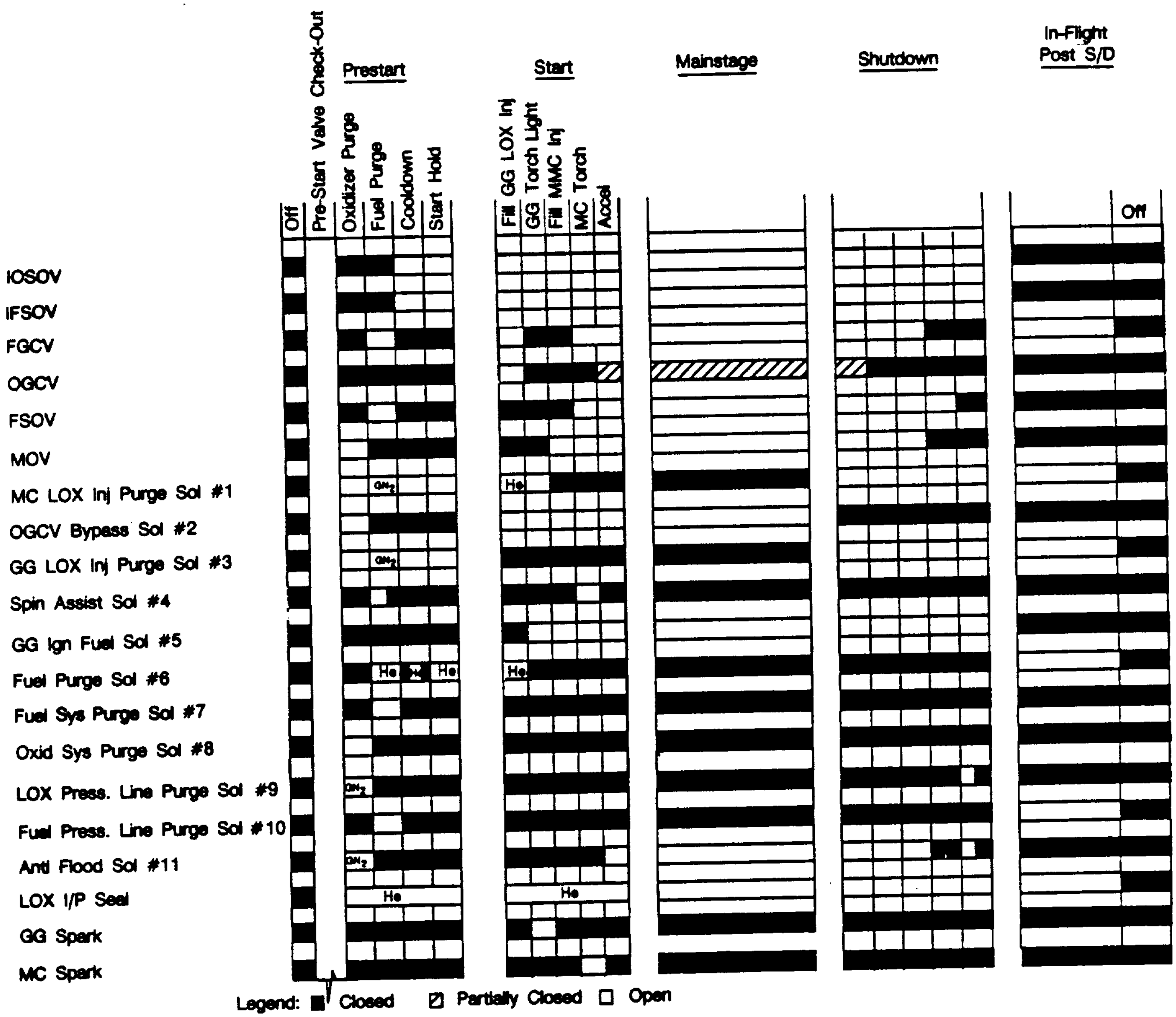

Figure 3.1.6-5. Valve Sequencing Accomplished With Timed Logic

3.1.6.5 Pumps Cooldown

The turbopumps are cooled to cryogenic temperatures by liquid hydrogen and liquid oxygen supplied through the vehicle inlet lines. Other than activating purge flows no control valve sequencing is required by the engine.

3.1.6.6 Start

The engine start is a timed sequence process using a LO_2 lead for both the gas generator (GG) and main chamber (MC). In the LO_2 lead concept GG and MC fuel is delayed until the injector volumes are filled and liquid oxygen flow is established. This results in a smooth start and eliminates the potential temperature spikes and combustion instability associated with two phase LOX injector flow.

Helium is introduced to the GG via the GG fuel injector simultaneously purging any oxygen from the fuel injector and providing helium spin up assist to improve start repeatability and help in achieving the 5 second start requirement. Figure 3.1.6-2 shows the valve scheduling and thrust building characteristic during the start. Thrust buildup rates can be tailored to meet the start requirement by modifying the GG valve start schedule. A bypass valve (OGCVBP) is used to provide LO_2 starting flow prior to opening the GG valves. Fuel rich torches are used for ignition of both the gas generator and main chamber. The use of a fuel rich torch is compatible with safe, fast and reliable ignition when an LO_2 lead start is used.

3.1.6.7 Main Stage

Main stage engine operation is an open-loop process. Analysis has shown that an open-loop control concept can be used to meet the ± 3.0% thrust, and mixture ratio requirement, at constant inlet pressure, once the engine is trimmed at the 712K thrust point during the acceptance test. Engine mixture ratio and gas generator mixture ratio are remotely trimmed during engine acceptance testing by trimming the full open position of the MOV and FGCV respectively.

3.1.6.8 Shutdown

Shutdown is performed by scheduling the propellant valves closed. The OGCV and the OGCVBP are closed first to power down the turbopumps. The MOV and the FGCV are then closed. The FSOV, which shuts off all fuel flow to the engine, is closed last, thus completing the shutdown sequence.

The gas generator and main chamber LO_2 injector purge solenoid valves are opened when the shutdown signal is received from the vehicle. Check valves are included to prevent backflow into the purge lines. When LO_2 injector pressure drops below the checked helium supply pressure the helium purge flow will commence. This flow purges any LO_2 trapped downstream of the OGCV and MOV after they are closed.

Predicted characteristics of an engine shutdown from 712K thrust level are shown in Figure 3.1.6-3.

3.1.6.9 Post Shutdown

Fuel downstream of the fuel shutoff valve (FSOV) is purged out through the main chamber and fuel gas generator control valve (FGCV). Fuel upstream of the fuel shutoff valve (FSOV) and oxygen upstream of the main oxidizer valve (MOV), oxidizer gas generator control valve (OGCV) and OGCV Bypass is allowed to percolate back to the propellant tanks.

3.1.7 Engine Configuration and Integration

3.1.7.1 Derivative STBE Gas Generator Engine Assembly

The arrangement of the external configuration of the engine was based on consideration of accessibility for routine component inspections, removals and replacements. Figures 3.1.7-1 and -2 show the side and top views of the engine assembly and its major components.

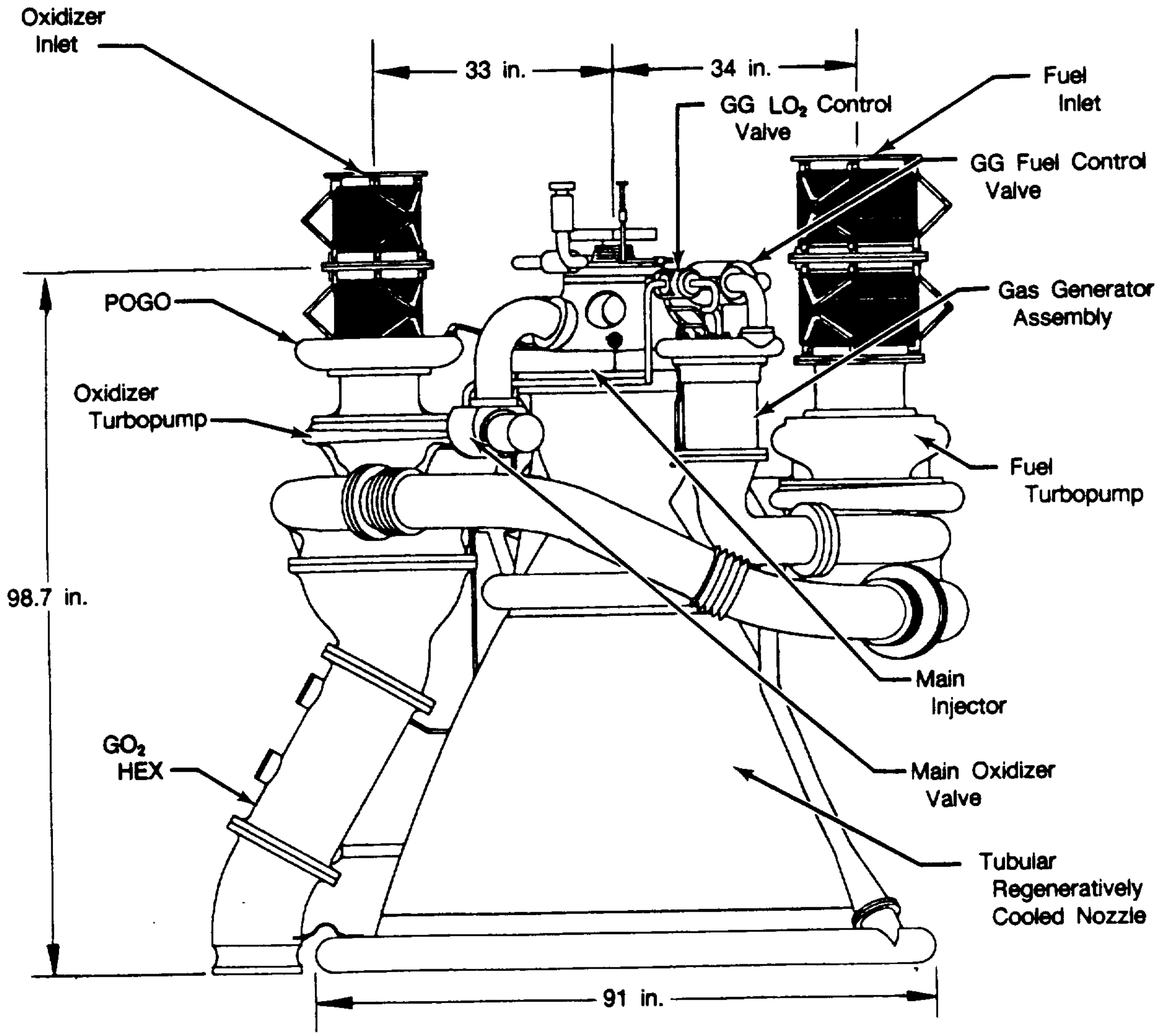

Figure 3.1.7-1. STBE Derivative Gas Generator Engine Assembly — Side View

Turbopumps are oriented on a vertical axis and cooldown recirculation valves have been eliminated, leading the way to cooldown by percolation. Engine propellant inlets accommodate engine gimballing through the use of scissor bellows mounted directly to the pump inlets. A toroidal shaped POGO accumulator has been incorporated between the LO_2 pump inlet and the scissors bellows. The engine thrust vectoring gimbal is incorporated into the main injector thrust structure. The gimbal design is based on a ball and socket feature with a central through-pin which restrains torsional movement. A teflon impregnated fiberglass fiber woven fabric between the gimbal ball and main injector socket is used as a friction reduction medium to permit engine gimballing. Gas generator/turbine exhaust is ultimately dumped overboard through the GO_2 heat exchanger and nozzle.

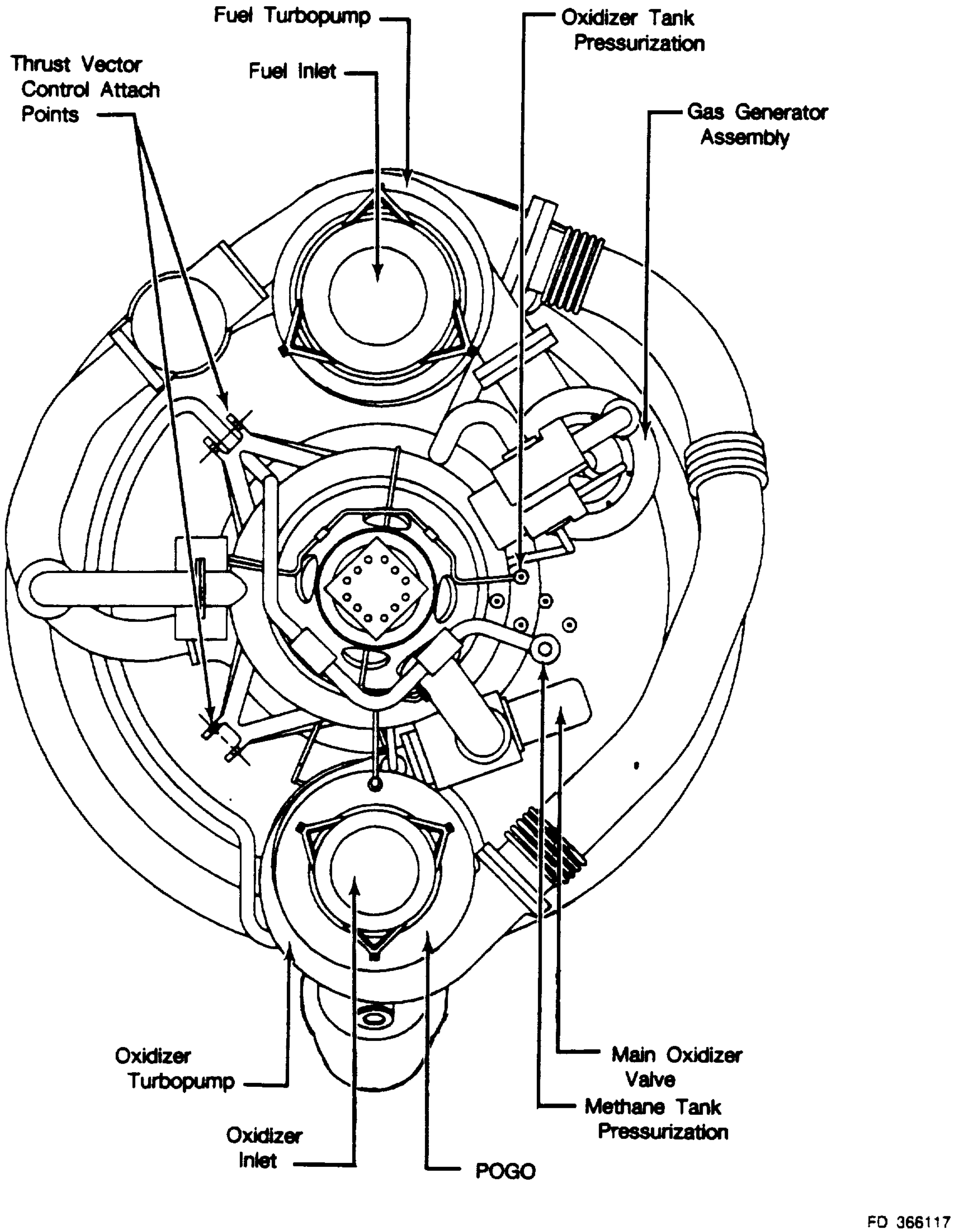

Figure 3.1.7-2. STBE Derivative Gas Generator Engine Assembly — Top View

All pneumatic and electrical interfaces are located at the engine interface plane, similar to the SSME.

3.1.7.2 Flex Joints

The baseline ALS engine designs use four types of flexible flow ducting joints, bipod stabilized bellows inlet ducts, internally restrained bellows joints, externally restrained bellows joints, and unrestrained compression joints.

The baseline designs which do not use boost pumps result in pump inlets located 33 and 34 inches from the gimbal centerline. Bipod stabilized bellows inlet ducts were selected due to their lower cost and lighter weight when compared to SSME type wraparound articulated ducts. To accommodate the large axial and angular deflections resulting from the 12-degree square pattern gimbaling requirements, the number of bellows convolutions and convolution height were iterated to obtain sufficient flexibility for deflection capability while retaining adequate bellows axial stiffness to prevent squirm due to internal pressure.

The resulting inlet ducts consist of two three-ply Inconel 718 10-inch long bellows with 25 one-inch tall convolutions per duct. These ducts have been designed for a nominal gimbal capability of ± 6 degrees. However, analyses have been conducted to evaluate increases in gimbal capability up to ± 12 degrees. Stabilizing linkages separate the two bellows to prevent buckling of the duct assembly. Excursion limiting stops are included on the stabilizing links to prevent overdeflection of the bellows. Preliminary analysis indicates that at the 12-degree gimbal level, this configuration meets stress criteria but has little margin for bellows squirm. Future analysis is required to optimize the bellows configuration to minimize the stress levels and to provide additional squirm margin. Vibration analysis is needed to evaluate the potential for flow induced vibration resulting from the vortex shedding phenomena. Some internal bellows damping effect is anticipated due to the three-ply bellows construction. Internal flow guides will be considered, however, their use is complicated by the large axial deflections resulting from the 12-degree gimbaling.

Approximately two degrees of torsional deflection is required on the duct during maximum gimbaling. A low spring-rate bellows torsional spring will likely have to be incorporated in the duct assembly to prevent overstressing of the bellows or pump inlet housings.

An additional consideration is the large percentage volume change which occurs in the duct during severe gimbaling. If the resulting flow pulse in the LO_2 duct causes significant thrust oscillations, the use of pressure-volume compensating ducts as used on the F-1, or wraparound articulated ducts will have to be evaluated.

In the event that the bipod stabilized ducts prove unsatisfactory for gimbal capability greater than ± 6 degrees, after future analysis, wrap around articulated ducts will likely be chosen for the inlet or intermediate pressure ducts. Three types of gimbal joints were studied for possible inclusion in these ducts: internal ball strut joints, externally pinned joints, and external ball race joints.

The internal ball strut joint, shown in Figure 3.1.7-3, contains a ball and socket joint supported by struts in both halves of the joint which guides the joint angulation. A bellows encloses the entire joint assembly. The bellows must carry torsion loads which can cause bellows column buckling when deflected. The main advantage of this configuration is its light weight and small volume. The small envelope size allows it to be easily vacuum jacketed for use in liquid hydrogen ducts. Its simplicity allows it to be the most inexpensive joint while achieving a high degree of reliability. Due to its low torsional load carrying capabilities, its use will likely be limited to hydrogen ducting since the higher density of methane or LO_2 may produce excessive torsional loads on the joint under g-loading. This joint is also used as the baseline for intermediate pressure hot gas flow ducting between turbopump turbines to allow thermal growth in the hot lines.

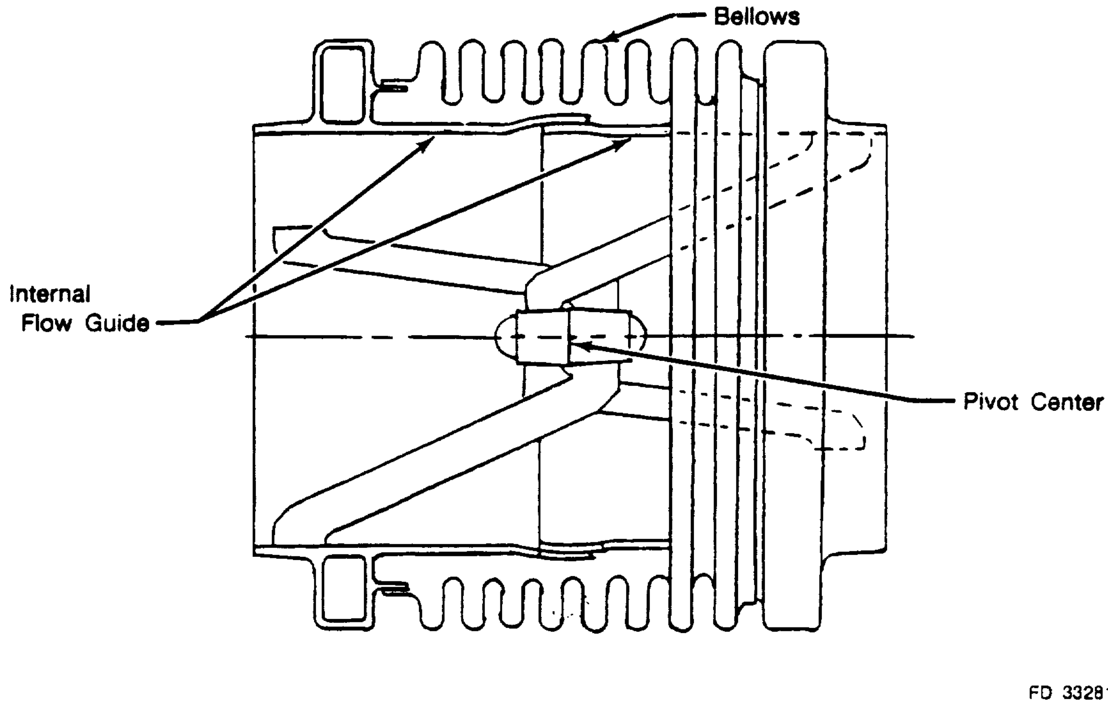

Figure 3.1.7-3. Internal Ball Strut Ducting Gimbal

The externally pinned joint, shown in Figure 3.1.7-4, uses a universal joint on the outside of the joint which carries all torsional loads. The flow bellows are not subject to torsion loads. The main advantage of this configuration is its low pressure drop due to the lack of obstructions in the flowpath. This configuration has the highest torsional loading capability of the three candidates making it the choice for ducting the higher density fluids. The joint is marginally heavier than the internal ball strut joint and displays similar reliability levels.

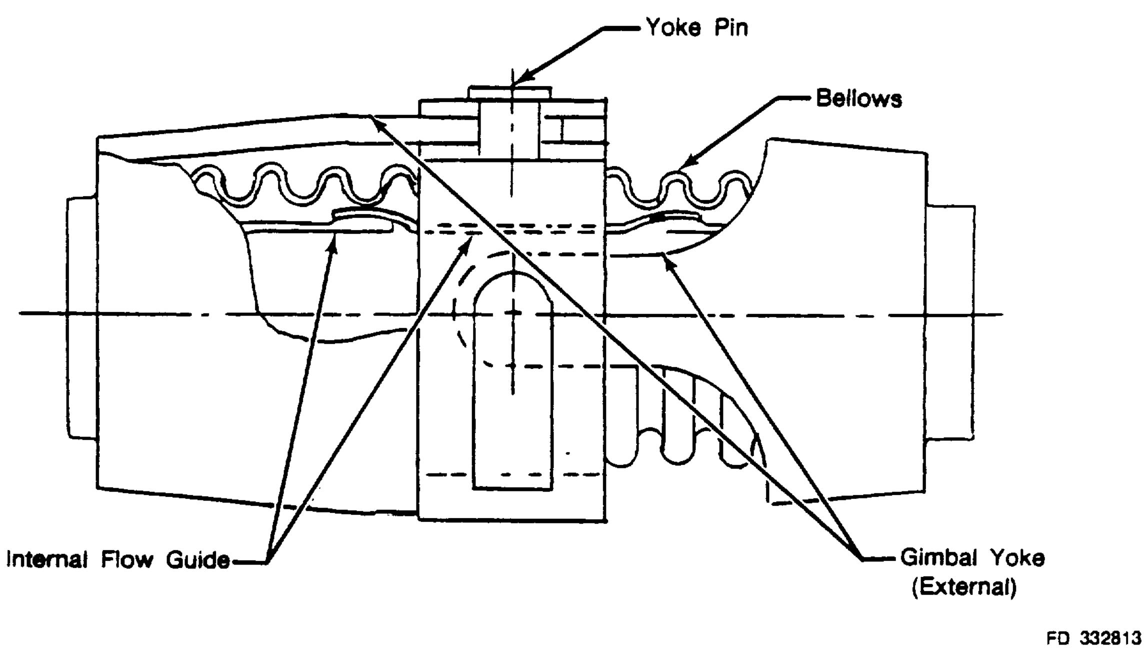

Figure 3.1.7-4. Externally Pinned Ducting Gimbal

The external ball race, shown in Figure 3.1.7-5, is fastened on the upstream side of the joint to a spherical shell and the downstream side of the joint is fastened in an inner spherical shell. The two shells are separated by ball bearings to reduce friction and are pressure-loaded together to guide the bellows during deflection. This design configuration is the heaviest and provides the lowest angulation levels of the candidate joints and therefore has been eliminated from further consideration. As the bellows carries torsional loads in this design it also has limited torsional capability.

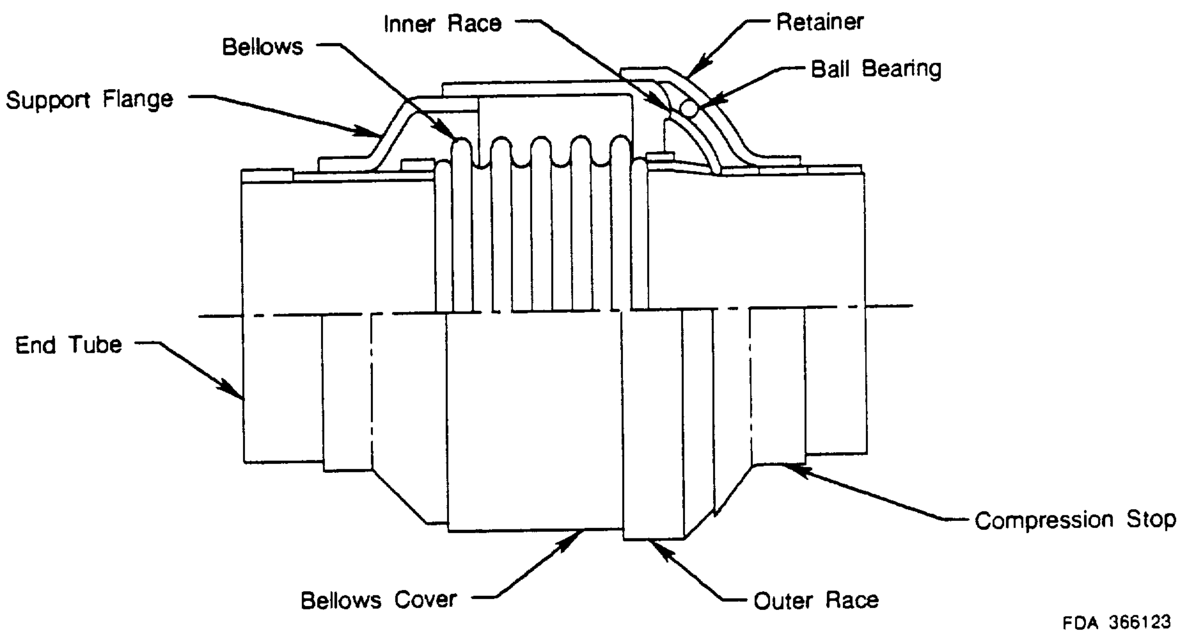

Figure 3.1.7-5. External Ball Race Ducting Gimbal

Both of the baseline joints are capable of ±15 degrees of angulation which should be adequate to allow 12-degree engine gimbaling in wraparound duct configurations. Internal flow liners will maintain acceptable flow characteristics and minimize flow induced bellows vibration.

Unrestrained bellows joints are used in the low pressure turbine exhaust to allow thermal expansion of the ducts. Due to the low pressures, the axial loads transmitted into the mating duct and manifold are low enough to not require a restrained bellows. Care must be taken in designing these ducts to ensure efficient load transfer from the bellows into the surrounding hardware. If the operational deflections of the engine components are large enough, these ducts may be installed in an opposite deflection (loaded) position to allow the duct to move toward a neutral and lower stress position during operation.

3.1.7.3 GO_2 Heat Exchanger

The STBE GO_2 heat exchanger, which is common with the STME GO_2 HEX, has been designed to provide gaseous oxygen to the oxygen tank for tank pressurization. The GO_2 heat exchanger uses the gas generator exhaust duct flow as the heat source to vaporize the liquid oxygen as shown in Figure 3.1.2-1. The heat exchanger surface is provided by three Haynes 214 stainless steel tubes wrapped in parallel around the gas generator exhaust duct. The gas generator exhaust duct wall is made of beryllium copper with trip-strip roughened walls to enhance the heat transfer. The tubes are packed in powdered copper to structurally isolate the tubes from the duct wall, while providing a good heat transfer medium. This heat exchanger design eliminates the possibility of accidental mixing of the oxygen and gas generator exhaust flow, thereby eliminating a category 1 failure mode.

The GO_2 heat exchanger will require three ⅜-inch diameter tubes 50-feet long, wrapped around the 12-inch duct. The tubes have 0.015-inch thick walls and are separated from one another by 0.055 inch, requiring a total duct length of 1.5 feet. Figure 3.1.7-6 diagrammatically presents the GO_2 heat exchanger geometry. The GO_2 heat exchanger has been thermally analyzed for the STBE engine operating point of 100 percent thrust. The oxygen flow rate is predicted to be 5.0 lbm/sec. The heat exchanger has been designed to supply 850 R oxygen to the tank. Figure 3.1.7-6 also summarizes the predicted heat exchanger performance.

3.1.7.4 Engine Performance

The STBE derivative gas generator engine system performance was determined using the accepted JANNAF methodology. Vacuum specific impulse was calculated separately for both the main chamber nozzle and the GG nozzle. Overall engine performance was calculated by mass weighing the main chamber flow performance with the GG flow performance. Table 3.1.7-1 summarizes main chamber and GG performance parameters at the design thrust level of 644,898 pounds sea level.

During this study program, detailed aerothermal analyses were made to predict component performance levels. Results of these analyses were incorporated into a steady state power balance model of the complete engine. A simplified flow schematic is presented in Figure 3.1.7-7 with key operating parameters noted for the design thrust level. Table 3.1.7-2 defines performance of the individual components and their operating environments for the derivative engine at design power level.

3.1.7.5 Engine/Vehicle Interface Requirements

All engine physical interfaces meet ALS ICD specifications. The fuel and oxidizer inlet ducts are configured on a 180-degree spacing and are 34 and 33 inches from the gimbal centerline respectively. The engine assembly could be converted to a 90-degree pump inlet spacing if a benefit to the vehicle is found to exist. A review of the vehicle contractors current vehicle cluster configurations indicates better access to the turbopumps when installed on the vehicle. As the engine maintenance concept evolves, module and LRU location of the engine assembly will be reviewed. Currently, hydrodynamic design has assumed that the inlet ducts are free of bends and are the same diameter as the pump inlet for at least five pipe diameters upstream of the inducer. As vehicle configurations stabilize, the sensitivity of the pump designs to inlet flow perturbations will be more fully addressed.

In addition to the propellant inlets, four additional fluid interfaces exist on the baseline engines: the two propellant tank pressurization flows, a nitrogen and a helium supply for engine purges. SSME interface locations were used for these fluid interfaces on current baseline ALS engine designs. Significant flexibility in the location of these lines exists to respond to vehicle requirements.

Nitrogen is required only during ground purges. Helium is required for the engine start system and for inflight purges and post shutdown purges. For those engine recovery concepts which involve sea recovery, an additional purge of the turbopump turbine cavities and bearing compartments prior to water impact through shipboard recovery is required to prevent corrosive sea air from being drawn into the turbopumps as the hot turbine structures cool. Vehicle considerations will likely guide the decision to use either nitrogen or helium for this purge. Additional refinement and quantification of the turbopump cavity volumes are required to quantify the flowrate requirements for all purges.

STBE GO_2 Heat Exchanger
Thermal and Flow Performance

Gas Generator Exhaust Duct

Thrust - %	100
Temperature - R	1550
Pressure - psia	289.1
Flow Rate - lbm/sec	185.4

Oxygen

Flow Rate - lbm/sec	5.0
Inlet Temperature - R	179
Exit Temperature - R	850
Inlet Pressure - psia	3229
Exit Pressure - psia	2889

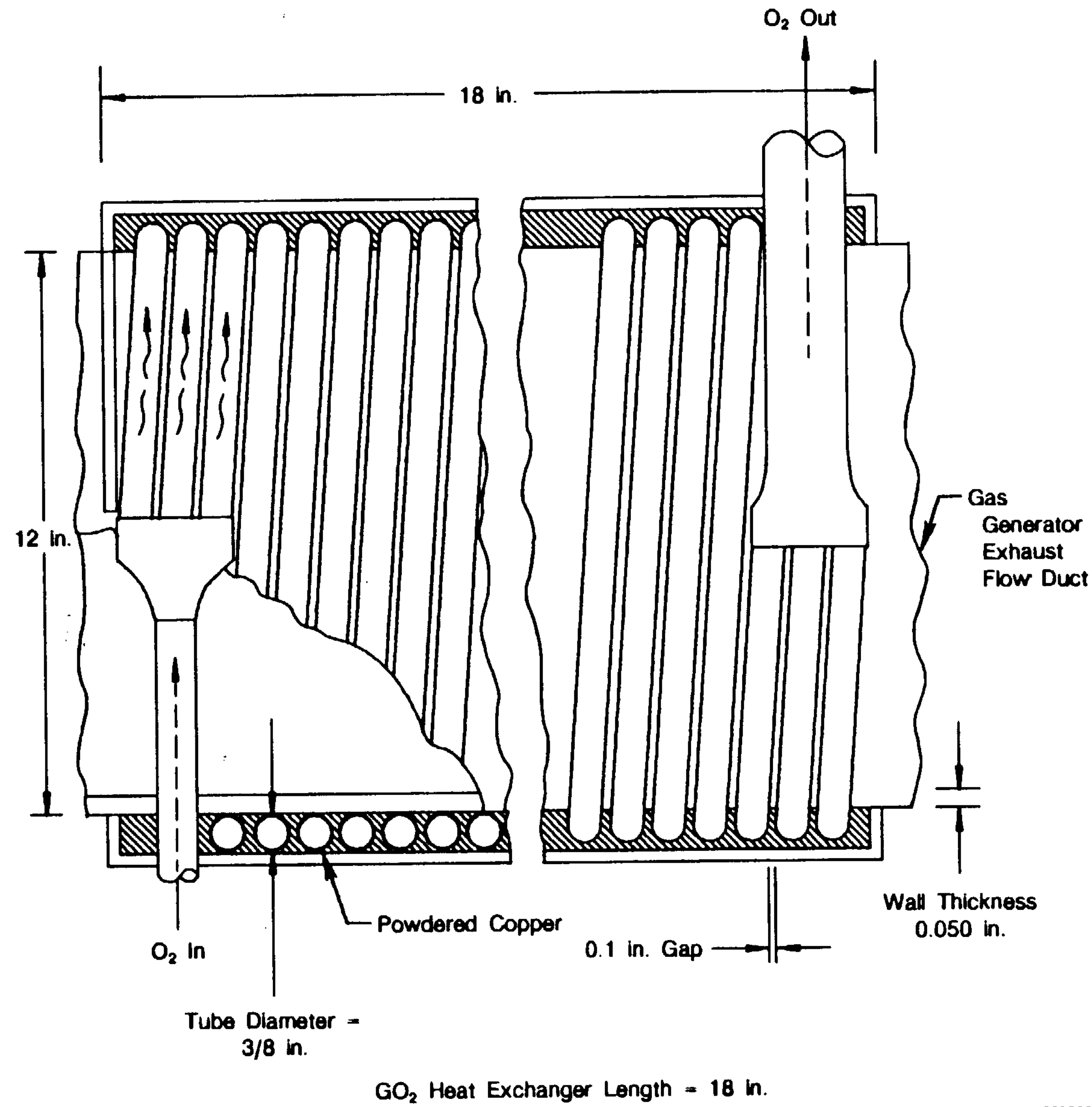

Figure 3.1.7-6. STBE Derivative Gas Generator GO_2 HEX Geometry and Performance Data

Table 3.1.7-1. STBE Derivative Gas Generator Engine Performance — Design Power Level

	Design Power Level Main Chamber	Gas Generator
Pressure — psia	2250	221.6
Mixture Ratio	3.48	0.301
Nozzle Area Ratio	28	5
Flow Rate — lb/sec	1993	185
Vacuum Thrust — lb	679922	31901
Vacuum I_{sp} — sec	342.9	172.4
	Overall Engine	
Vacuum Thrust — lb	711,823	
Vacuum Del. I_{sp} — sec	328.4	
S.L. Thrust — lb	644,898	
S.L. I_{sp} — sec	297.5	

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The proposed method of supplying vehicle electrical power is a vehicle mounted generator coupled to an auxiliary turbine driven by the fuel tank pressurization flow. Pressure drop across the generator turbine lowers fuel tank pressurization flow to the 500 psi level downstream of the turbine. A conceptual design has been completed which would supply 25 kW DC power per engine, or 75 kW total in a three-power engine cluster. Growth margin exists to increase the 25 kW level if vehicle requirements increase. This concept removes the generator from the engine assembly to reduce gimbaled mass, lowering actuator loads. This approach is attractive since it is compact and does not require a separate hydrazine APU system as on the shuttle. Use of this system would require the tank pressurization flow to be continuous, not pulsed. If hydraulically operated thrust vectoring actuators are selected, an electrically driven hydraulic pump would be required in conjunction with this system.

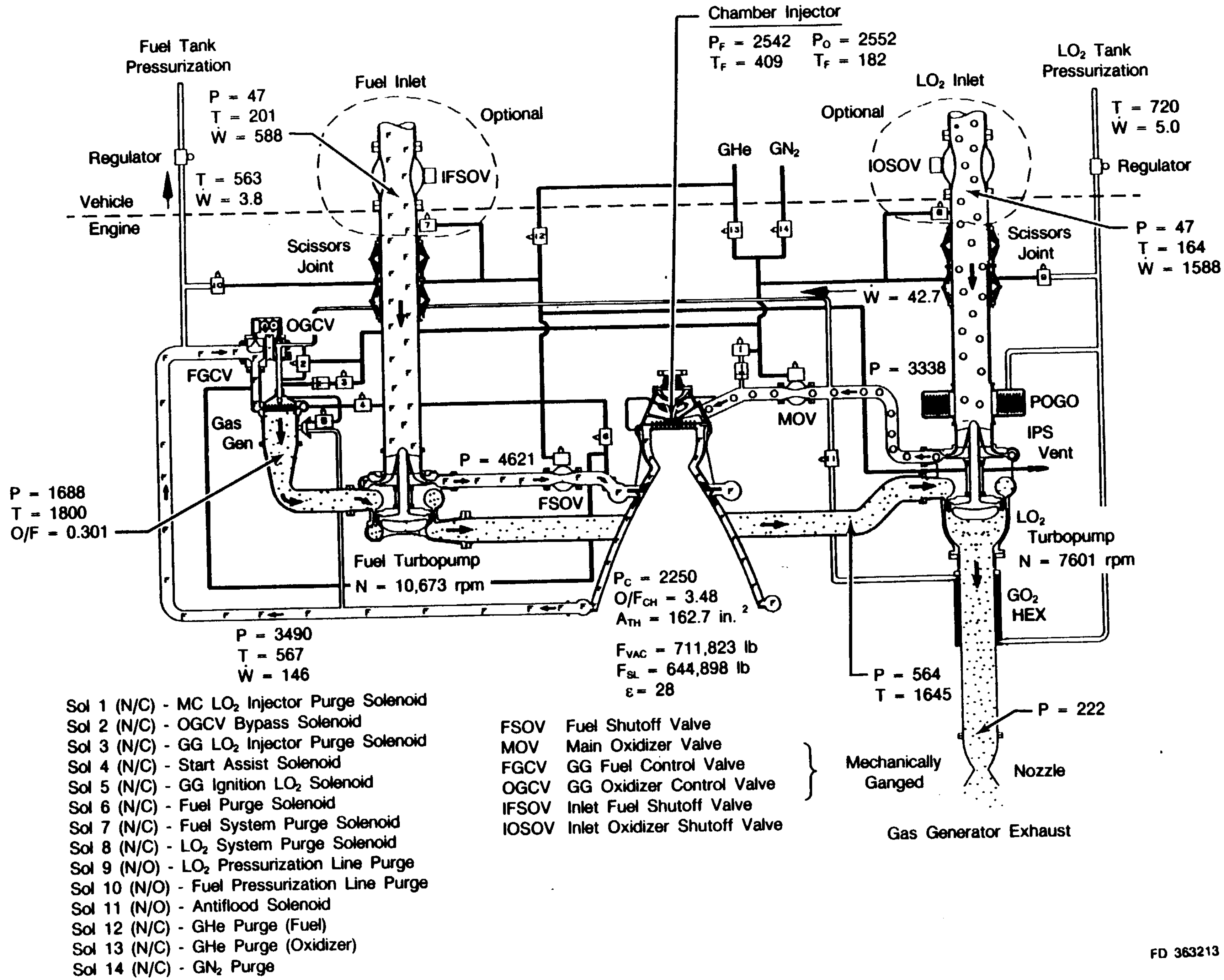


Figure 3.1.7-7. STBE Derivative Gas Generator Engine Operating Characteristics at DPL

Table 3.1.7-2. *STBE Derivative Gas Generator Engine Performance — Design Power Level*

```
****************************************
*            PRATT & WHITNEY           *
* GAS GENERATOR CYCLE OFF-DESIGN DECK  *
*          STBE ENGINE STUDY           *
****************************************
```

ENGINE PERFORMANCE

VACUUM THRUST	711823.
SEA LEVEL THRUST	644898.
VACUUM IMPULSE	328.35
SEA LEVEL IMPULSE	297.48
TOTAL ENGINE INLET FLOW RATE	2176.6
OVERALL ENGINE MIXTURE RATIO	2.70

ENGINE HEAT TRANSFER

CHAMBER COOLANT DP	1806.
CHAMBER COOLANT DT	177.
CHAMBER Q	66563.
NOZZLE COOLANT DP	534.
NOZZLE COOLANT DT	333.
NOZZLE Q	41724.

CHAMBER PERFORMANCE

PRESSURE	2250.0
TEMPERATURE	6601.7
THRUST	679922.
IMPULSE	342.90
FLOW RATE	1982.9
THROAT AREA	162.71
NOZZLE AREA RATIO	28.
MIXTURE RATIO	3.48
NOZZLE EFFICIENCY	0.965
CSTAR EFFICIENCY	0.980

GAS GENERATOR PERFORMANCE

PRESSURE	1687.5
TEMPERATURE	1800.0
THRUST	31901.
IMPULSE	172.44
FLOW RATE	185.0
MIXTURE RATIO	0.301
NOZZLE EFFICIENCY	0.970
NOZZLE GAS CONSTANT	97.2
NOZZLE GAMMA	1.177
NOZZLE AREA	88.7

ENGINE STATION CONDITIONS

* FUEL SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	47.0	201.0	588.3	123.1	26.40
1ST STAGE EXIT	2325.1	216.0	588.3	145.4	26.55
MAIN PUMP EXIT	4621.3	230.3	588.3	167.2	26.74
FSOV INLET	4506.9	231.0	588.3	167.2	26.68
FSOV EXIT	4451.2	231.3	588.3	167.2	26.64
CHAM/COOL INLET	4369.8	231.8	442.2	167.2	26.60
CHAM/COOL EXIT	2563.5	408.8	442.2	317.8	15.79
CH INJ INLET	2542.0	408.6	442.2	317.8	15.75
NOZ/COOL INLET	4024.2	233.8	146.0	167.2	26.40
NOZ/COOL EXIT	3490.4	566.7	146.0	453.0	10.42
TANK PRESS OUT	3249.7	563.0	3.8	453.0	9.93
TANK PRESS IN	47.0	421.4	3.8	453.0	0.17
FGCV INLET	3249.7	563.0	142.3	453.0	9.93
FGCV EXIT	2385.1	545.3	142.3	453.0	7.91
GG INJ INLET	2276.1	542.4	142.3	453.0	7.63

* OXIDIZER SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	47.0	164.0	1588.3	61.6	71.17
MAIN PUMP EXIT	3338.3	178.8	1588.3	72.8	71.74
GOX HEX IN	3228.9	179.3	5.0	72.8	71.58
TANK PRESS IN	47.0	720.0	5.0	275.4	0.22
MOV INLET	3228.9	179.3	1540.6	72.8	71.58
MOV EXIT	2647.4	181.7	1540.6	72.8	70.70
CH INJ INLET	2552.5	182.0	1540.6	72.8	70.55
OGCV INLET	2880.3	180.7	42.7	72.8	71.05
OGCV EXIT	2668.2	181.6	42.7	72.8	70.73
GG INJ INLET	2434.7	182.5	42.7	72.8	70.37

* GAS GEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW
FUEL TURB INLET	1532.7	1800.0	185.0
FUEL TURB EXIT	646.7	1655.5	185.0
LOX TURB INLET	563.9	1645.4	185.0
LOX TURB EXIT	289.1	1550.3	185.0
NOZZLE INLET PRES	221.6		

Table 3.1.7-2. *STBE Derivative Gas Generator Engine Performance — Design Power Level (Continued)*

```
**************************************
*           PRATT & WHITNEY          *
* GAS GENERATOR CYCLE OFF-DESIGN DECK *
*          STBE ENGINE STUDY         *
**************************************
```

TURBOMACHINERY PERFORMANCE DATA

* FUEL TURBINE *

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.774	0.759
HORSEPOWER	19000.	17763.
SPEED (RPM)	10673.	10673.
S SPEED	35.2	44.3
S DIAMETER	1.84	1.54
MEAN DIAMETER (IN)	19.12	19.10
VEL.RATIO (ACTUAL)	0.47	0.48
MAX TIP SPEED	918.	941.
BLADE HEIGHT	0.58	1.10
AN SQUARED	39.7	75.2
EFFECTIVE AREA	14.07	21.17
PRES.RATIO (T/T)	1.54	1.54
GAS CONSTANT (FT)	97.20	
GAMMA	1.1626	

* FUEL PUMP *

	STAGE ONE	STAGE TWO
EFFICIENCY	0.713	0.729
HORSEPOWER	18570.	18192.
SPEED (RPM)	10673.	10673.
NPSH (FT)	177.7	12461.5
SS SPEED	27007.	1130.
S SPEED	910.	905.
HEAD (FT)	12373.	12405.
DIAMETER (IN)	18.28	18.28
TIP SPEED (FT/SEC)	852.	852.
VOL FLOW	10002.	9945.
HEAD COEF	0.5396	0.5410
FLOW COEF	0.0728	0.0724

* LOX TURBINE *

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.774	0.699
HORSEPOWER	12630.	12637.
SPEED (RPM)	7601.	7601.
S SPEED	50.9	55.4
S DIAMETER	1.16	1.06
MEAN DIAMETER (IN)	18.90	18.77
VEL.RATIO (ACTUAL)	0.40	0.40
MAX TIP SPEED	680.	715.
BLADE HEIGHT	1.60	2.77
AN SQUARED	54.9	94.4
EFFECTIVE AREA	38.67	50.39
PRES.RATIO (T/T)	1.36	1.43
GAS CONSTANT (FT)	97.05	
GAMMA	1.1697	

* LOX PUMP *

EFFICIENCY	0.756
HORSEPOWER	25267.
SPEED (RPM)	7601.
NPSH (FT)	62.4
SS SPEED	37052.
S SPEED	1037.
HEAD (FT)	6618.
DIAMETER (IN)	18.91
TIP SPEED (FT/SEC)	628.
VOL FLOW	10017.
HEAD COEF	0.5406
FLOW COEF	0.0821

* VALVE DATA *

STATION	DELP	AREA	FLOW	%DELP/P
FUEL SHUT OFF VLV	55.7	22.83	588.3	1.24
FUEL GG VALVE	864.5	2.803	142.3	26.60
MAIN OXID VALVE	581.6	11.30	1540.6	18.01
LOX GG VALVE	212.0	0.521	42.7	7.36

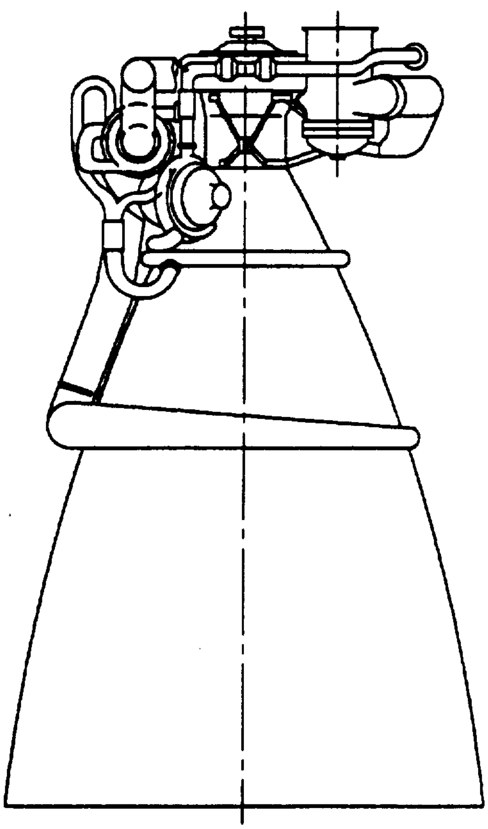
* INJECTOR DATA *

STATION	DELP	AREA	FLOW	%DELP/P
FUEL GG INJ	588.6	3.960	142.3	25.86
FUEL CH INJ	292.0	13.19	442.2	11.49
LOX GG INJ	747.2	0.279	42.7	30.69
LOX CH INJ	302.5	15.78	1540.6	11.85

3.2 UNIQUE STBE LO_2/CH_4 GAS GENERATOR CYCLE ENGINE

3.2.1 Unique Gas Generator Engine Design Evolution

The unique LO_2/methane gas generator engine cycle study was initiated in the first quarter of 1988. The first engine design is shown in Figure 3.2.1-1 with engine characteristics. This engine was a 625,000-pound (625K) sea level fixed thrust with the design point at 688K sea level thrust. The first bipropellant unique engine design incorporated all of the STME/STBE low cost design and manufacturing concepts. These concepts are listed in Table 3.2.1-1. This was the prime expendable concept when the tripropellant was the prime reusable concept. Reliability predictions, unit production costs, and the impact on life cycle cost were evaluated for the bipropellant, expendable 625K fixed thrust sea level engine design during the first quarter of 1988. The results of these evaluations are presented in P&W Interim Report FR-19691-3.



Gas Generator Cycle

Propellants	LO_2/CH_4
Mixture Ratio	3.04
Chamber Pressure	2044 psia
Thrust - Vacuum	717,500 lb
- Sea Level	625,000 lb
Specific Impulse - Vacuum	340.1 sec
- Sea Level	296.2 sec
Nozzle Area Ratio	35
Length	145 in.
Diameter	90 in.
Weight	7014 lb

FD 359995

Figure 3.2.1-1. STBE Unique Gas Generator Cycle Engine — 625K Sea Level Thrust

During the second quarter of 1988, the LO_2/methane bipropellant engine concept was refined to include growth capability to 750K sea level thrust with some hardware changes. This engine design and its major characteristics are shown in Figure 3.2.1-2. Several design and analytical trade studies were conducted to substantiate the engine design. The major studies conducted were a boost pump trade study and a mixture ratio trade study.

Table 3.2.1-1. Design Changes To Reduce Fabrication Costs

- Simple Axial Inlet Turbopumps
- Removed MCC Igniter From Acoustic Liner for Simplification of Chamber
- Simplified MCC Coolant Channel Geometry
- Eliminated Expensive/Complex Wrap-Around Flex Lines
- Cast Turbopump Housings
- Changed to Lower Cost Materials Wherever Possible
- Equiaxed Turbine Blades
- Cast Oxygen and Fuel Pump Impellers
- Cast Gas Generator and MCC Injector Elements and Divider Plate
- Cast Chamber With Electroplate Nickel Closeout and Bicast Structural Jacket
- Filament Wound Shell on Tubular Nozzle
- Formed Tubular Nozzle

R19691/39

As the bipropellant common engine study began to emerge as the focus of STBE efforts, the engine design did not undergo further study until the fourth quarter of 1988 and continued through the first quarter of 1989. This engine assembly design and overall characteristics are presented in Figure 3.2.1-3. This 750K engine incorporates all of the low-cost concepts as previously discussed except that the turbopumps are mounted vertically. The following paragraphs refer to the design definition of this 750K sea level thrust engine shown in Figure 3.2.1-3, with low cost design and manufacturing features and vertical turbopumps.

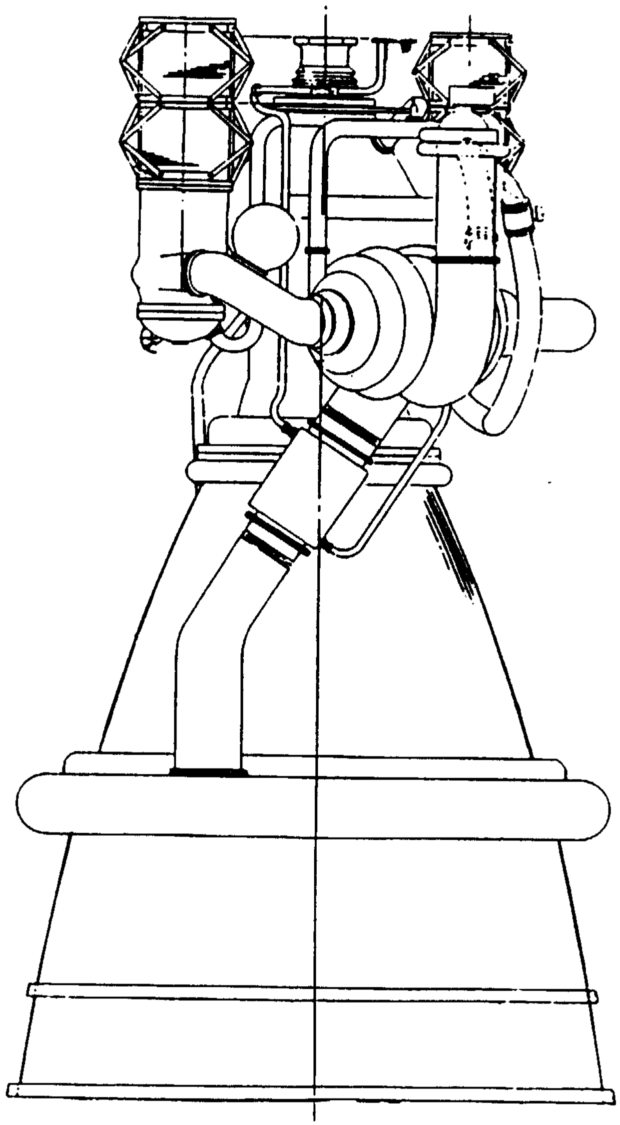
3.2.2 Engine Cycle

The candidate unique LO_2/CH_4 STBE configuration studied during the Phase A' extension is a gas generator cycle with liquid oxygen and liquid methane as propellants. This engine operates at a main chamber pressure of 2396 psia at the design power level (DPL) of 750,000 pounds thrust and has the capability of running at a nominal power level (NPL) of 625,000 pounds thrust. The engine has a fixed nozzle with an area ratio of 35:1 and delivers 305 seconds of sea level specific impulse at DPL. Figure 3.2.1-3 presents selected engine characteristics at the rated power level.

3.2.2.1 Flowpath Description

A simplified flow schematic, showing the major flowpaths and components for the STBE, is presented in Figure 3.2.2-1.

Liquid oxygen enters the engine at a net positive suction head (NPSH) level, supplied by the vehicle, sufficient for the high-speed high-pressure methane pump; thus boost pumps are not required for this system.

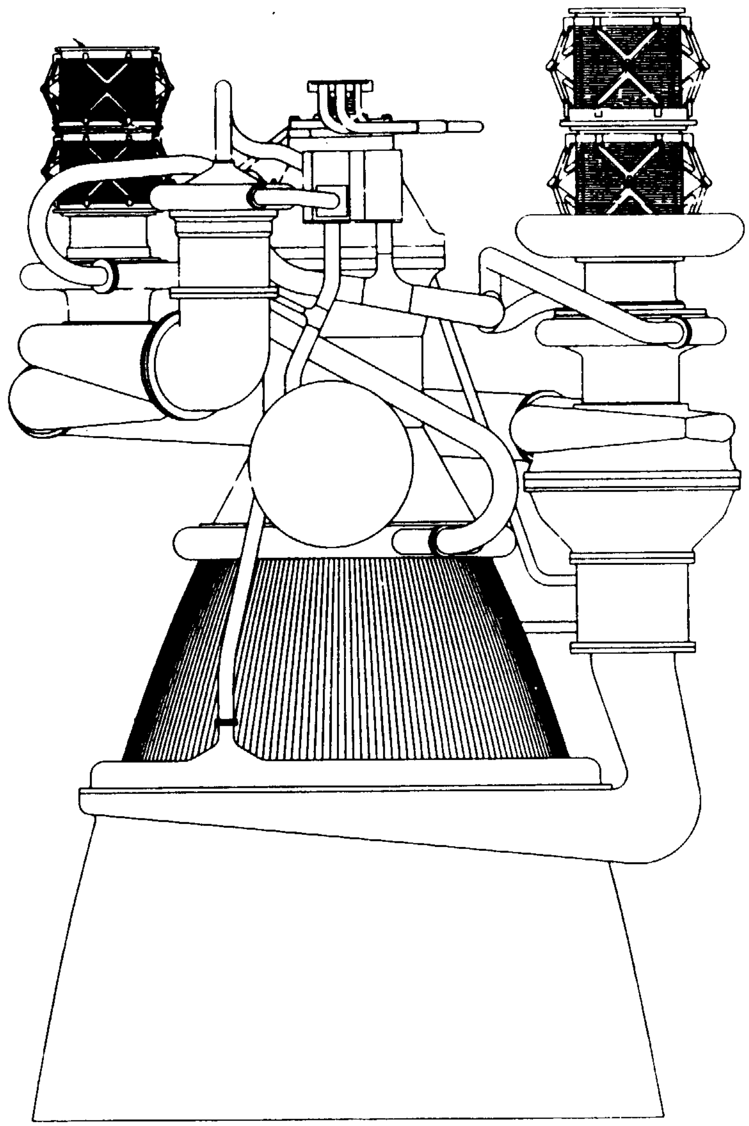


Gas Generator Cycle

Propellants	LO_2/CH_4
Mixture Ratio	3.0
Chamber Pressure	2250 psia
Thrust - Vacuum - Sea Level	709,300 lb 625,000 lb
Specific Impulse - Vacuum - Sea Level	339.0 sec 298.7 sec
Nozzle Area Ratio	35
Diameter	92 in.
Length	161 in.
Weight	6476 lb

FD 359996

Figure 3.2.1-2. STBE Unique Gas Generator Cycle Engine — 750K Sea Level Thrust — Growth Capability

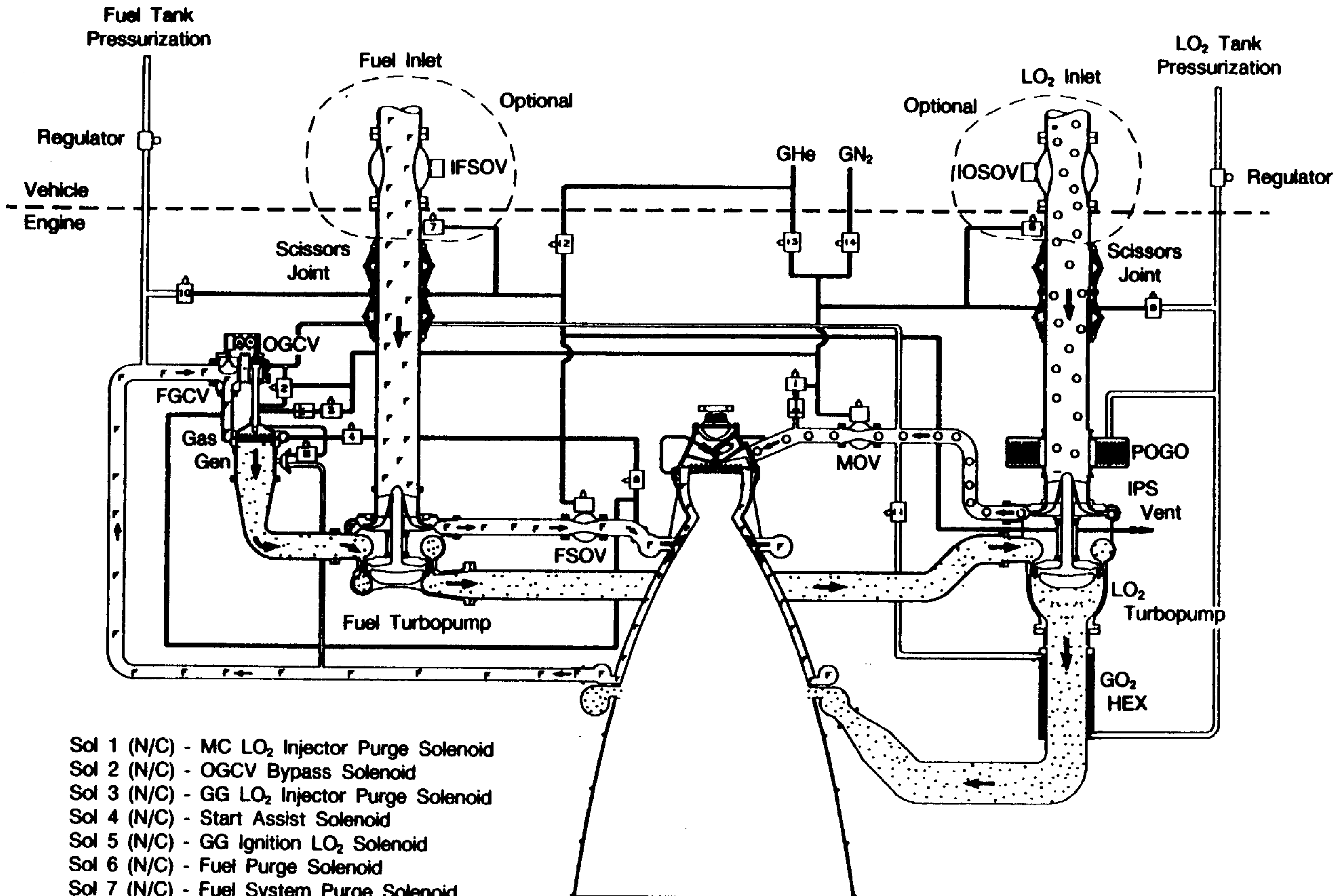


Performance

Thrust - Vacuum	841,390 lb
- Sea Level	750,000 lb
Chamber Pressure	2396 psia
Mixture Ratio	3.0
Specific Impulse - Vacuum	342 sec
- Sea Level	305 sec
Weight	XX lb
Thrust-To-Weight	XX
Area Ratio	35
Length	89 in.
Diameter	142 in.

FD 359977

Figure 3.2.1-3. STBE Unique Gas Generator Characteristics at Design Power Level



Sol 1 (N/C) - MC LO_2 Injector Purge Solenoid
Sol 2 (N/C) - OGCV Bypass Solenoid
Sol 3 (N/C) - GG LO_2 Injector Purge Solenoid
Sol 4 (N/C) - Start Assist Solenoid
Sol 5 (N/C) - GG Ignition LO_2 Solenoid
Sol 6 (N/C) - Fuel Purge Solenoid
Sol 7 (N/C) - Fuel System Purge Solenoid
Sol 8 (N/C) - LO_2 System Purge Solenoid
Sol 9 (N/O) - LO_2 Pressurization Line Purge
Sol 10 (N/O) - Fuel Pressurization Line Purge
Sol 11 (N/O) - Antiflood Solenoid
Sol 12 (N/C) - GHe Purge (Fuel)
Sol 13 (N/C) - GHe Purge (Oxidizer)
Sol 14 (N/C) - GN_2 Purge

FSOV Fuel Shutoff Valve
MOV Main Oxidizer Valve
FGCV GG Fuel Control Valve } Mechanically Ganged
OGCV GG Oxidizer Control Valve }
IFSOV Inlet Fuel Shutoff Valve
IOSOV Inlet Oxidizer Shutoff Valve

FD 363210

Figure 3.2.2-1. Simplified Flow Schematic for STBE Unique Gas Generator Cycle Engine

At the design power level, the methane pump operates at 17,181 rpm to provide the methane pressure level of 5195 psia required by the cycle. From the pump exit, the methane flows through the fuel shutoff valve to a split manifold at the inlet of the coolant passages. From the split manifold, 81.2 percent of the methane is used to regeneratively cool the milled channel, copper alloy main chamber from an area ratio of 5.48:1 back to the injector face. The remaining methane flow is used to cool the tubular stainless steel nozzle from an area ratio of 5.48:1 down to an area ratio of 35:1. This methane then flows through the fuel gas generator control valve and is injected into the gas generator to combust with some of the oxygen to provide power for the high pressure turbomachinery.

The high-pressure oxidizer pump operates at 6,787 rpm to provide the oxygen pressure level of 3046 psia required by the cycle at the design power level. From the pump exit, approximately 98.3 percent of the oxygen flow is routed through the main oxidizer control valve and is injected into the main chamber. The remainder of the oxygen flows through the oxygen gas generator control valve before being injected into the gas generator.

The high-pressure, high-temperature (2281 psia/1800°R at DPL) gas of the gas generator provides the power to drive the high-pressure propellant pumps. The hot gas is initially expanded through the methane turbine and is subsequently routed to a second turbine which powers the oxygen pump. From the oxidizer turbine discharge, the flow enters a heat exchanger where energy is extracted to vaporize the oxygen being provided for tank pressurization. The turbine exhaust gas is then expanded through an area ratio of 5:1 to atmospheric pressure, providing additional thrust to the overall engine output.

3.3 COMMON STBE LO_2/CH_4 GAS GENERATOR CYCLE ENGINE

3.3.1 Engine Design Evolution

The common STBE/STME Gas Generator Cycle Engine design has evolved from a 388,000-pound (388K) sea level design thrust, very common engine to a higher thrust with considerable part commonality but minimal performance penalty to the STME. The common engine concepts were based upon the following guidelines during conceptual design studies:

- Use unique STME hardware wherever possible for both the STBE and STME engines
- Where unique STME engine hardware cannot be used for both engines, design the most common piece of hardware, while minimizing performance debit to the STME engine, i.e., main injector
- Where a common piece of hardware could not be used, (such as the main combustion chamber), design a new part for the booster engine application, and use the unique STME design for the main engine application.

Four separate engine designs resulted from this commonality philosophy in the STME and STBE programs:

1. Unique STME
2. Common STME (similar to the unique STME with slight performance, cost, and weight penalties)

3. Unique STBE

4. Common STBE (significant performance penalty when compared to the unique STBE).

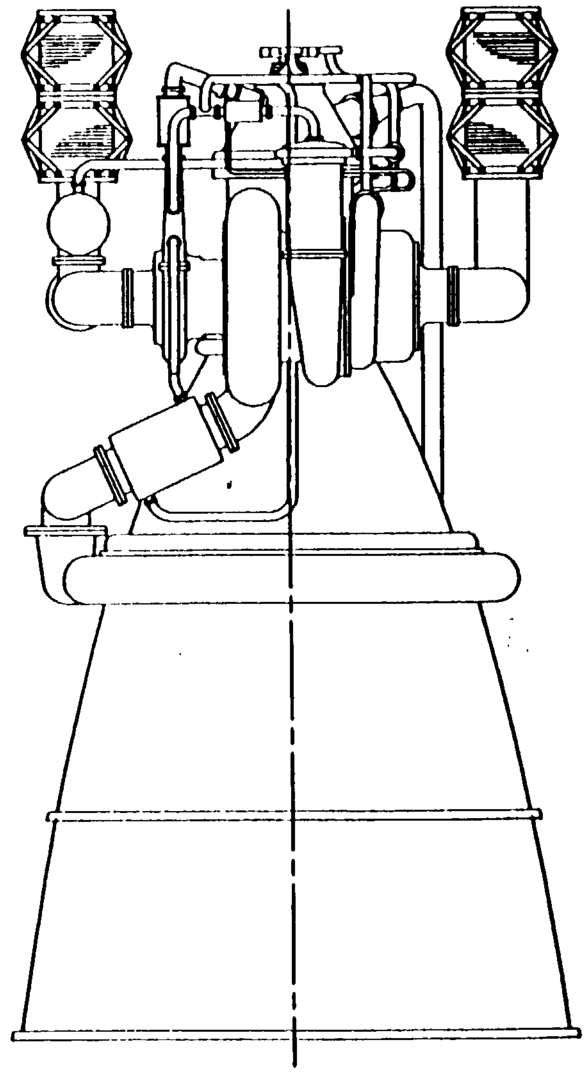
The first common engine design, in which a common main combustion chamber was used, resulted in a low-thrust booster engine design. The engine external assembly and characteristics are shown in Figure 3.3.1-1, for operation with both LO_2/H_2 and LO_2/CH_4 as propellants.

This low-thrust level in the STBE that resulted from a most common STME/STBE engine proved to be unacceptable for use as an ALS booster engine. Therefore, design changes required to produce a 750K sea level thrust STBE engine resulted in a new design (but common) main combustion chamber and major pump housings. The engine assembly design and major characteristics are shown in Figure 3.3.1-2. Due to the higher fuel system pressures in the STBE cycle, the common chamber, controls, pump housings, and large ducting lines imposed a large weight penalty on the STME. The common STME thrust-to-weight ratio was approximately 56.5:1, while the unique STME thrust-to-weight ratio was 85:1. The results of this study prompted P&W to back off on the second guideline, engine commonality, and design separate, unique main combustion chambers, major turbopump housings, and large ducting lines and controls for each engine. This engine assembly and major characteristics are shown in Figure 3.3.1-3. This engine design was further refined to minimize performance, cost, and weight penalties to the STME, while maximizing part commonality between the two common engines and maintaining STBE thrust at an acceptable level of 635K sea level thrust. A comparative summary of the major engine components for the Unique 580K STME design, the Common 580K STME design, and the 635K STBE engine is presented in Table 3.3.1-1.

3.3.2 Engine Cycle

The STME/STBE common gas generator configuration, studied during the Phase A' contract, uses liquid oxygen and liquid hydrogen as propellants for the STME engine; while liquid oxygen and liquid methane is the fuel for the STBE engine. This engine operates at a main chamber pressure of 2250 psia at the Design Power Level (DPL) of 580K lbf vacuum thrust for the STME; and 635K lbf sea level thrust for the STBE. The STME engine has a fixed nozzle with an area ratio of 62:1 that delivers 440.0 seconds of vacuum specific impulse at DPL. The STBE engine uses an STME nozzle that is truncated at an area ratio of 35:1 and delivers 295.4 seconds of sea level specific impulse. Figure 3.3.1-3 presents selected engine characteristics at the design power level for the STME/Common STBE engine.

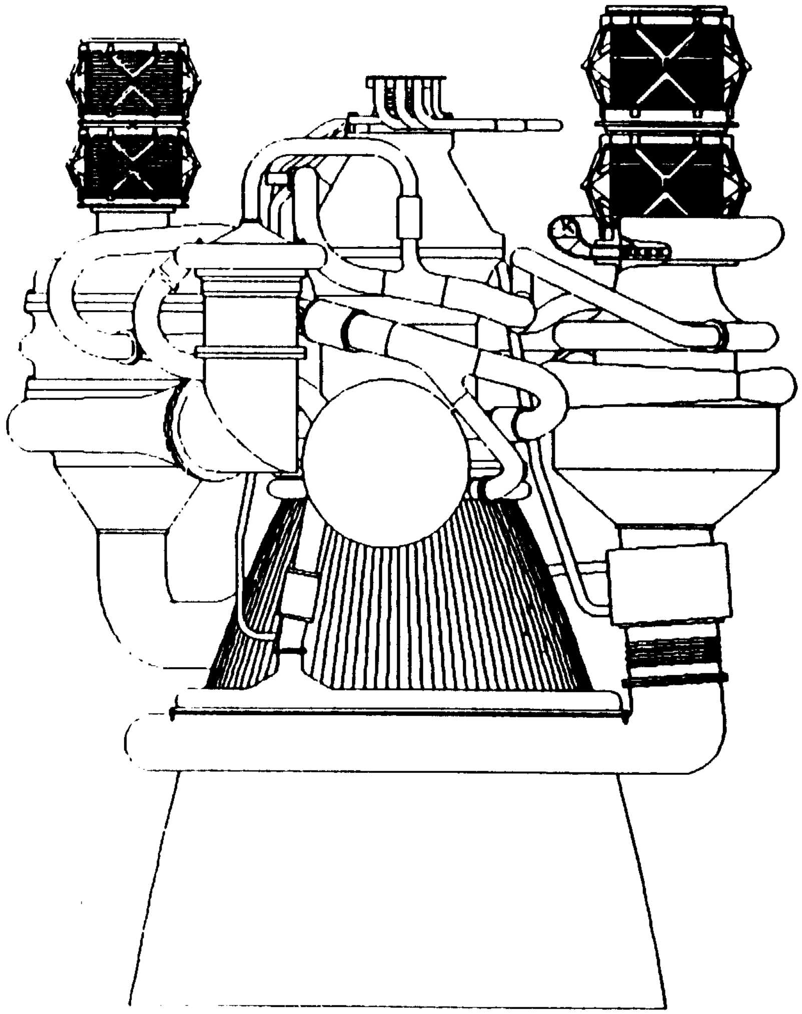
The benefit of engine commonality is the reduction of manufacturing costs. The common hardware between the STME/STBE Common gas generator engines are as follows: pumps, turbines, gas generator, combustor, nozzle, igniter, injectors, controls, GO_2 heat exchanger, LO_2 POGO suppressor, LO_2 vent and main valves. However, some modifications had to be made and are: restaggering of STBE turbine blades; truncation of the STBE nozzle at an area ratio of 35:1; change fuel orifices in the STBE injector; and some software changes in the engine controller.



Propellants	LO_2/H_2	LO_2/CH_4
Mixture Ratio	6.0	3.0
Chamber Pressure	2678 psia	2250 psia
Thrust - Vacuum	435,000 lb	349,700 lb
- Sea Level	340,700 lb	312,700 lb
Specific Impulse - Vacuum	448.1 sec	344.6 sec
- Sea Level	351.0 sec	308.0 sec
Nozzle Area Ratio	80	35
Diameter	96 in.	64 in.
Length	168 in.	92 in.
Weight	4731 lb	TBD lb

FD 359981

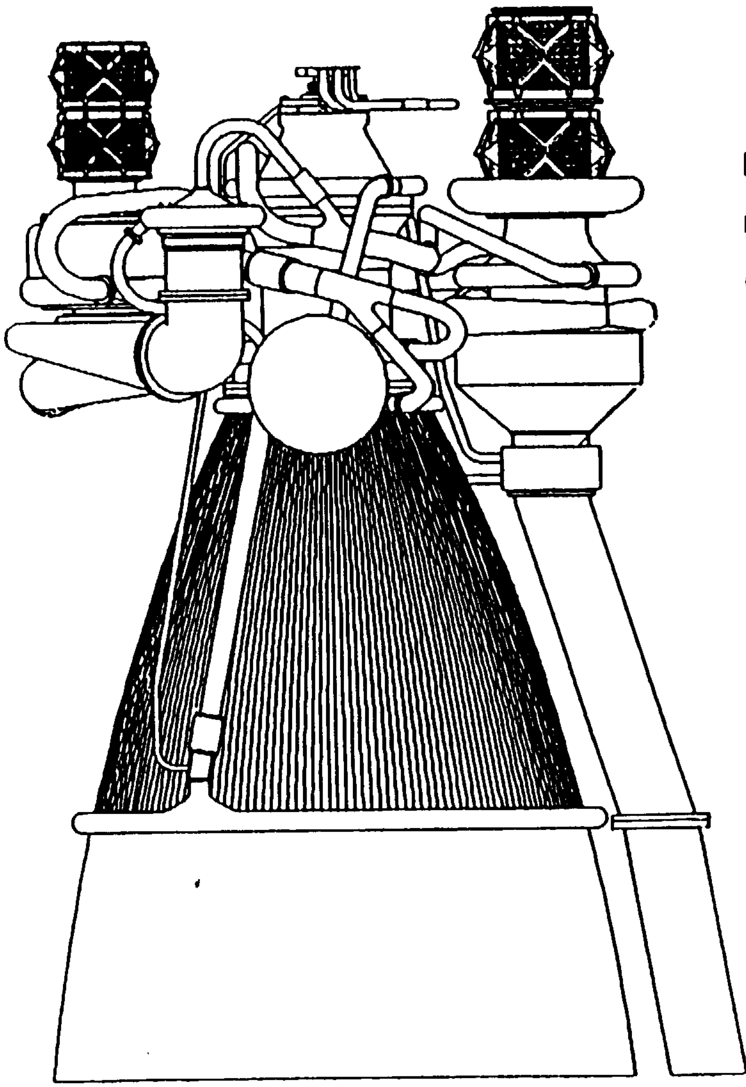
Figure 3.3.1-1. STBE Gas Generator Common Engine at Normal Operating Conditions



Propellants	LO_2/H_2	LO_2/CH_4
Mixture Ratio	6.0	2.70
Chamber Pressure	2250 psia	3159 psia
Thrust - Vacuum	580,000 lb	816,225 lb
- Sea Level	462,750 lb	750,024 lb
Specific Impulse - Vacuum	436.5 sec	337.3 sec
- Sea Level	348.2 sec	309.9 sec
Nozzle Area Ratio	62	35
Diameter	104 in.	76 in.
Length	174 in.	133 in.
Weight	10,254 lb	10,766 lb

FD 359982

Figure 3.3.1-2. STBE Gas Generator Common Engine at Design Operating Conditions



Propellants	LO_2/H_2	LO_2/CH_4
Mixture Ratio	6.0	3.0
Chamber Pressure	2250 psia	2250 psia
Thrust - Vacuum	580,000 lb	719,035 lb
- Sea Level	460,921 lb	635,008 lb
Specific Impulse - Vacuum	440.1 sec	334.5 sec
- Sea Level	349.7 sec	295.4 sec
Nozzle Area Ratio	62	35
Diameter	102 in.	91 in.
Length	179 in.	136 in.
Weight	TBD lb	TBD lb

FD 359983

Figure 3.3.1-3. STBE Gas Generator Common Engine at Design Operating Conditions

Table 3.3.1-1. *STME/STBE Common Engine and STME H_2/O_2 Unique Engine Comparison — Common Hardware Evaluation*

Component	*Unique STME*	*Common GG H_2 Engine*	*Common GG CH_4 Engine*
Fuel System			
Pump	Unique	New Design	Same as H_2
Vent Valve	Unique	Same as CH_4	New Design
Shutoff Valve	Unique	Same as CH_4	New Design
Coolant Valve	Unique	Same as CH_4	New Design
GG Control Valve	Unique	Same as CH_4	New Design
Gas Generator	Unique	New Design	Same as H_2
Turbine	Unique	New Design	Reblade From H_2, Same Housings, etc.
Oxidizer System			
POGO Suppressor	Common	Same as STME	Same as STME
Vent Valve	Unique	Same as CH_4	New Design
Main Valve	Unique	Same as CH_4	New Design
Heat Exchanger	Common	Same as STME	Same as STME
Turbine	Unique	New Design	Reblade From H_2, Same Housings, etc.
Chamber and Injector			
Injector	Unique	New Design	Same as H_2
Regeneratively Cooled Nozzle	Unique	New Design	Same as H_2
Film Cooled Nozzle	Unique	New Design	No Additional Nozzle
Igniter	Common	Same as STME	Same as STME, Modified Flow Control Orifices
Combustor	Unique	New Design	New Design With Acoustic Liner
Controls			
Instrumentation	Common	Same as STME	Same as STME
Engine Controller	Common	Same as STME	Same as STME Software Change
Engine Assembly			
Engine Ducting	50% Common	50% Same as STME	Same as H_2
Vehicle Interfaces	Common	Same as STME	Same as STME
Gimbal	Common	Same as STME	Same as STME

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3.3.2.1 Flow Path Description

A simplified flow schematic for the STME/STBE common engine is presented in Figure 3.3.2-1, showing the major flowpaths and components.

Liquid oxygen enters the engine at a net positive suction head (NPSH) level, supplied by the vehicle, sufficient for the high-speed high-pressure oxidizer pump. The fuel enters the engine at a NPSH level, again supplied by the vehicle, sufficient for the high-speed high-pressure fuel pump; thus boost pumps are not required for this system.

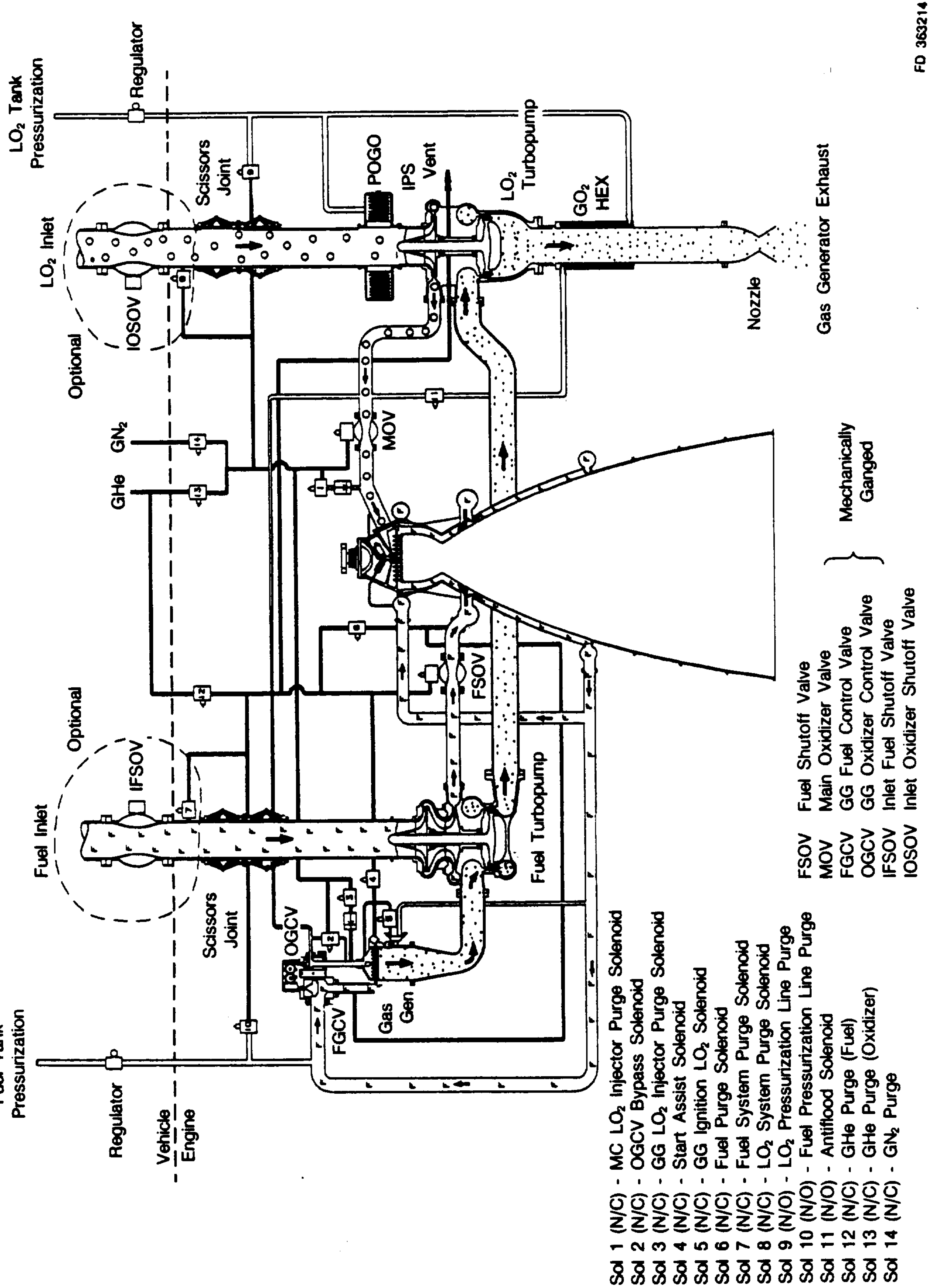


Figure 3.3.2-1. Simplified Flow Schematic for STME/STBE Common Gas Generator Cycle Engine

At the Design Power Level for the STME (STBE), the fuel pump operates at 21,660 rpm (10,478 rpm) to provide a fuel pressure level of 3456 psia (4710 psia) required by the cycle. From the pump exit, the fuel flows through the fuel shutoff valve to a split manifold at the inlet of the coolant passages. From the split manifold, 70.5 percent (45.5 percent) of the fuel is used to regeneratively cool the milled-channel copper alloy main chamber from an area ratio of 5.86:1 back to the injector face. Then it is routed directly into the injector manifold and then the main combustion chamber. The remaining fuel flow is used to cool the tubular stainless steel nozzle from an area ratio of 35:1. Subsequently, the nozzle cooling flow splits where 38.7 percent (37.5 percent) is supplied to the gas generator and the rest is routed on to the main thrust chamber. The fuel supplied to the gas generator control valve is injected into the gas generator to combust with some of the oxygen to provide power for the high pressure turbomachinery.

The high-pressure oxidizer pump operates at 6435 rpm (7500 rpm) to provide the oxygen pressure level of 2784 psia (3336 psia) required by the cycle at the Design Power Level. From the pump exit, approximately 98 percent of the oxygen flow is routed through the main oxidizer control valve and is injected into the main chamber. The remainder of the oxygen flows through the oxygen gas generator control valve before being injected into the gas generator.

The gas generator provides 1175 psia (2400 psia), 1800 R gas to drive the high-pressure propellant pumps. The hot gas is initially expanded through the fuel turbine and is subsequently routed to a second turbine which powers the oxygen pump. The turbine exhaust gas is then expanded through an area ratio of 5:1 to atmospheric pressure; thus providing additional thrust to the overall engine output.

3.4 UNIQUE STBE LO_2/RP-1 GAS GENERATOR CYCLE ENGINE

3.4.1 Engine Design Evolution

The LO_2/RP-1 STBE is a gas generator cycle engine with liquid oxygen and RP-1 as propellants. The engine design was initiated in the first quarter of 1988 and discussed in FR-19691-3 at 625K lb sea level thrust.

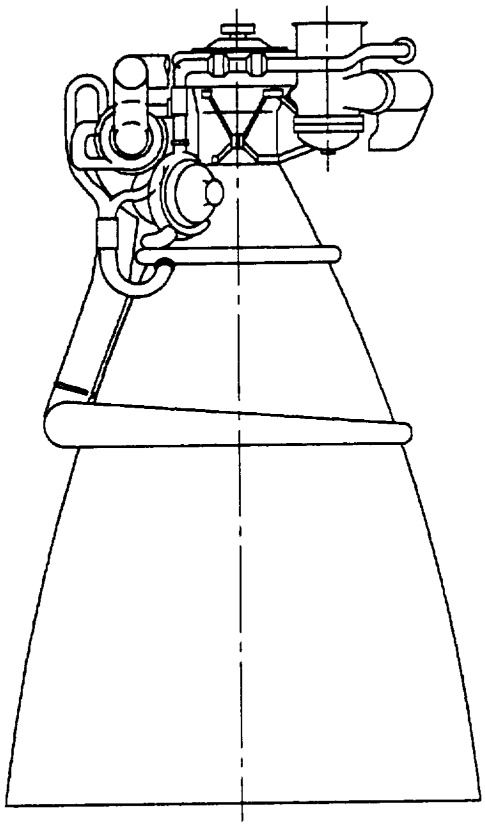
This engine study was continued to refine the LO_2/RP-1 gas generator engine design through the last quarter of 1988. The significant changes from the initial engine design were the increase in design thrust level to 750K lb sea level and the elimination of boost pumps due to the higher vehicle supplied NPSH. The engine assembly drawing and its major characteristics are shown in Figure 3.4.1-1.

3.4.2 Engine Cycle

The candidate STBE configuration studied during Phase A is a gas generator cycle with liquid oxygen and liquid RP-1 as propellants. This engine operates at a main chamber pressure of 1501 psia at the design power level (DPL) of 750K lb sea level thrust and has the capability of running at a nominal power level (NPL) of 625,000 pounds thrust. The engine has a fixed nozzle with an area ratio of 25:1 and delivers 274.6 seconds of sea level specific impulse at DPL.

3.4.2.1 Flow Path Description

A simplified flow schematic for the LO_2/RP-1 STBE is presented in Figure 3.4.2-1, showing the major flow paths and components.



Gas Generator Cycle

Propellants	LO_2/RP-1
Mixture Ratio	2.75
Chamber Pressure	1500 psia
Thrust - Vacuum	863,191 lb
- Sea Level	750,000 lb
Specific Impulse - Vacuum	316.0 sec
- Sea Level	274.6 sec
Nozzle Area Ratio	25
Diameter	99 in.
Length	149 in.
Weight	TBD lb

FDA 363206

Figure 3.4.1-1. STBE LO_2/RP-1 Gas Generator Engine Performance Characteristics at Design Power Level

Liquid oxygen and liquid RP-1 enters the engine at a net positive suction head (NPSH) level, supplied by the vehicle, sufficient for the high-speed, high-pressure pumps. No boost pumps are required for this system.

At the design power level, the RP-1 pump operates at 8,524 rpm to provide the RP-1 pressure levels of 2283 psia required by the cycle. From the pump exit, the RP-1 flow is split to cool the milled chamber and the tubular nozzle section separately. After cooling the nozzle, the gas generator flow is routed through a control valve and injected into the gas generator. The remainder of the nozzle coolant flow is mixed with the chamber coolant flow and is injected into the main chamber.

The high-pressure oxidizer pump operates at 5,645 rpm to provide the oxygen pressure level of 2091 psia required by the cycle at the design power level. From the pump exit, approximately 99 percent of the oxygen flow is routed through the main oxidizer control valve and is injected into the main chamber. The remainder of the oxygen flows through the oxygen gas generator control valve before being injected into the gas generator.

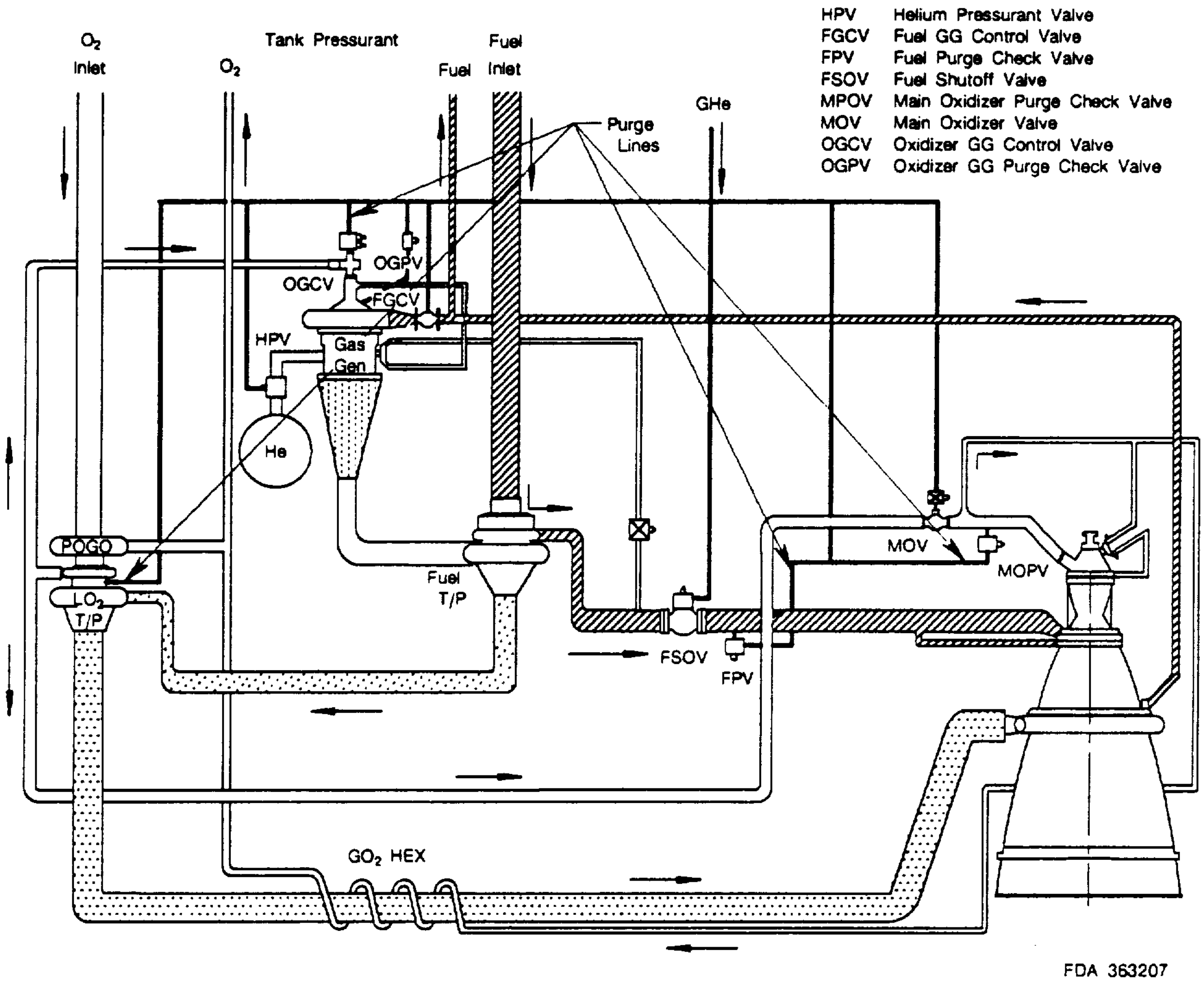


Figure 3.4.2-1. STBE LO_2/RP-1 Gas Generator Engine Simplified Flow Schematic

The high-pressure, high-temperature (1401 psia/1800 R at DPL) gas of the gas generator provides the power to drive the high-pressure propellant pumps. The hot gas is initially expanded through the RP-1 turbine and is subsequently routed to a second turbine which powers the oxygen pump. The turbine exhaust gas is then diverted down to the nozzle below the tubular nozzle section and is used to film-cool the remainder of the nozzle from an area ratio of 20:1 to the exit area of 25:1.

Liquid oxygen enters the engine at a net positive suction head (NPSH) level, supplied by the vehicle, sufficient for the high-speed high-pressure oxidizer pump. Liquid methane enters the engine at a NPSH level, again supplied by the vehicle, sufficient for the high-speed, high-pressure methane pump; thus boost pumps are not required for this system.

At the design power level, the RP-1 pump operates at 8524 rpm to provide the methane pressure level of 2283 psia required by the cycle. From the pump exit, the RP-1 flows through the fuel shutoff valve to a split manifold, 72.0 percent of the RP-1 is used to regeneratively cool the milled channel, copper alloy main chamber from an area ratio of 3.28 back to the injector face. The remaining RP-1 flow is used to cool the tubular, stainless steel nozzle from an area ratio of 3.28 down to an area ratio of 7.85:1. This RP-1 then flows through the fuel gas generator control valve and is injected into the gas generator to combust with some of the oxygen to provide power for the high-pressure turbomachinery.

The high-pressure oxidizer pump operates at 5645 rpm to provide the oxygen pressure level of 2091 psia required by the cycle at the design power level. From the pump exit, approximately 99.2 percent of the oxygen flow is routed through the main oxidizer control valve and is injected into the main chamber. The remainder of the oxygen flows through the oxygen gas generator control valve before being injected into the gas generator.

The high-pressure, high-temperature (1400 psia/1800 R at DPL) gas of the gas generator provides the power to drive the high-pressure propellant pumps. The hot gas is initially expanded through the RP-1 turbine and is subsequently routed to a second turbine which powers the oxygen pump. From the oxidizer turbine discharge, the flow enters a heat exchanger where energy is extracted to vaporize the oxygen being provided for tank pressurization. the turbine exhaust gas is then expanded through an area ratio of 25:1 to atmospheric pressure, providing additional thrust to the overall engine output.

3.4.2.2 Engine Operation

The engine will be preconditioned using liquid oxygen from the tank to soak the turbopump until it is sufficiently cooled. The oxidizer inlet valve will be opened, allowing liquid from the tank to flow down to the turbopump and letting any vapors percolate back up to the tank to be vented.

The engine start is a timed sequence process using an oxidizer lead for reliable soft propellant ignition. The oxidizer lead avoids hazardous buildup of unburned fuel in the combustor during the oxygen phase transition from gas to liquid. The transition occurs prior to fuel injection and the fuel is consumed immediately upon injection. Reliability of ignition is enhanced by the LO_2 lead because the transient mixture ratio during propellant filling includes the full excursion of ignitable mixture ratios from greater than 200 to less than one.

With the oxidizer lead sequence, the gas generator LO_2 injector is primed prior to opening the fuel shutoff valve to ensure liquid oxygen flow, eliminating turbine temperature spikes due to oxygen phase change. A helium spin assist is also used to initiate turbopump rotation before the fuel is introduced into the gas generator. During the start and shutdown, a small helium purge is used in the gas generator injector and main chamber injector to eliminate the danger of hot gas flow reversals during transient operation. Gas generator and main chamber ignition will be accomplished with dual electrical spark-excited torch igniters.

Main-stage engine operation is open-loop controlled. The fuel gas generator control valve (FGCV), the oxygen gas generator control valve (OGCV), and the main oxidizer valve (MOV), shown in Figure 3.4.2-1, are used to set the engine thrust and mixture ratio. Thrust and main chamber mixture ratio are set on the ground by trimming the MOV and OGCV respectively. The gas generator mixture ratio is set using the FGCV. All valves are operated by hydraulic actuators.

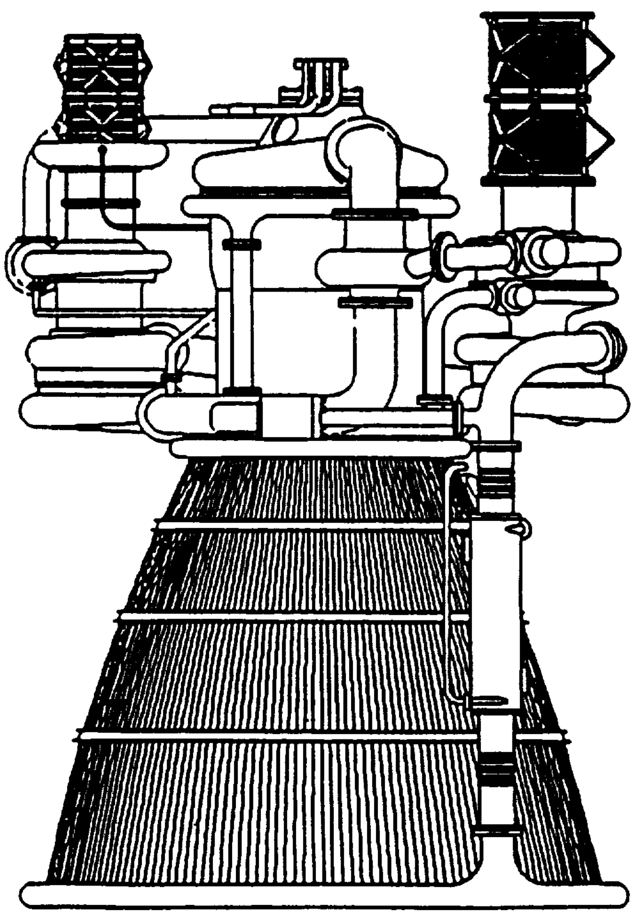
Engine acceleration is accomplished by a time-based scheduling of the valves to the commanded starting level (~ 20 percent power level). The acceleration to full thrust is also accomplished with open-loop valve schedules. Engine shutdown is accomplished using a time-based scheduling of the propellant valves. The OGCV is closed first to power down the turbopumps, then the MOV closes, followed by shutting off the RP-1 system.

In addition to a normal operational mode, the engine system is capable of shutdown resulting from detected problems or LO_2 starvation at the end of the burn duration.

3.5 DERIVATIVE LO_2/CH_4 SPLIT EXPANDER CYCLE ENGINE

3.5.1 Engine Design Evolution

The derivative, or modified split expander cycle engine study conceptual design was initiated as a result of the emerging need for a booster engine derived from the main engine. The 580K lbf vacuum thrust split expander main engine is designed for unique application to a core vehicle and delivers 433.9 seconds of vacuum specific impulse at the design power level using LO_2/H_2 as propellants. By utilizing as much hardware as possible, a derivative of this engine is designed to power a booster vehicle using LO_2/CH_4 as propellants. Both engines are presented in Figure 3.5.1-1 with overall engine characteristics.



Propellants	H_2/LO_2	CH_4/LO_2
Mixture Ratio	6.0	3.5
Chamber Pressure - psia	896	734
Thrust - Vacuum - sec	580,000	706,500
- Sea Level - sec	436,187	600,032
Specific Impulse - Vacuum - sec	433.9	327.7
- Sea Level - sec	326.3	278.3
Nozzle Area Ratio	28	13.5
Diameter - in.	116	104
Length - in.	187	140
Weight - lb	5,084	6,193

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Figure 3.5.1-1. STBE Derivative Split Expander Engines at Design Conditions

The derivative engine studies conducted during the last quarter of 1988 and first quarter of 1989 showed that maximum part commonality to the unique STME Split Expander engine could be achieved only at low booster engine thrust levels in the 300-400K range. Since the minimum acceptable sea level thrust for a booster engine application is 600K lbf, several new components were designed for the booster engine. Detailed discussion of this engine is presented in the following paragraphs.

3.5.2 Engine Cycle

The derivative STBE is a split expander cycle with liquid oxygen and liquid methane as the propellants. It is a derivative of the STME LO_2/hydrogen engine, and is intended to utilize as many STME hardware components as possible. This engine operates at a main chamber pressure of 734 psia at a fixed thrust level (NPL) of 600K lbf. The nozzle area ratio is optimized, for a booster engine application, at 13.5:1 and results in a delivered sea level specific impulse of 328 seconds.

3.5.2.1 Flow Path Description

A simplified flow schematic for the derivative STBE showing only the major flow paths and components is presented in Figure 3.5.2-1. Liquid oxygen and methane enter the engine at a NPSH level, supplied by the vehicle, sufficient for the high-speed, high-pressure pumps. No boost pumps are required in these systems. At normal power level, the methane pump operates at 10953 rpm to provide a first-stage pump discharge pressure level of 2546 psia. From the first-stage pump exit, 57 percent of the flow is routed to the second stage of the methane pump. The second-stage pump discharge level is 5740 psia. From the second-stage pump exit, the methane is routed through the nozzle shutoff valve into a split manifold chamber/nozzle. This heated methane is then used to provide power to drive the propellant pumps. Ninety percent of the nozzle cooling flow is routed through the turbines. The warm (689 R) methane gas is initially expanded through the methane pump drive turbine and is subsequently routed to a second turbine that powers the oxygen pump. The turbine exhaust is then routed to a mixer, where it combines with the remainder of the methane flow, and is then injected into the main chamber. At normal power level, the oxidizer pump operates at 5014 rpm to provide an oxygen pressure level of 1224 psia. From the pump exit, the oxygen flow is routed through a control valve and injected directly into the main chamber.

3.5.2.2 Engine Operation

The engine start is a timed sequence process using an oxidizer lead for reliable soft propellant ignition. The oxidizer lead avoids hazardous buildup of unburned fuel in the combustor or on the pad, because all fuel is consumed immediately upon injection. Reliability of ignition is enhanced by the LO_2 lead because the transient mixture ratio during propellant filling includes the full excursion of ignitable mixture ratios from greater than 200 to less than one.

With the oxidizer lead sequence, the LO_2 injector is primed prior to opening the fuel shutoff valve to assure liquid oxidizer flow. During the start and shutdown, a small helium purge is used in the main chamber injector to eliminate the danger of hot gas flow reversals during transient operation. Main chamber ignition will be accomplished with an electrical, spark-excited, oxygen/methane torch igniter.

Engine operation is controlled by a timed sequence of the five valves: nozzle shut-off valve, (NSOV), jacket bypass valve, (JBV), fuel shut-off valve, (FSOV), turbine shut-off valve, (TBV), and main oxidizer valve (MOV) (Figure 3.5.2-1). Engine acceleration is accomplished by scheduling the valves on open-loop schedules to full thrust.

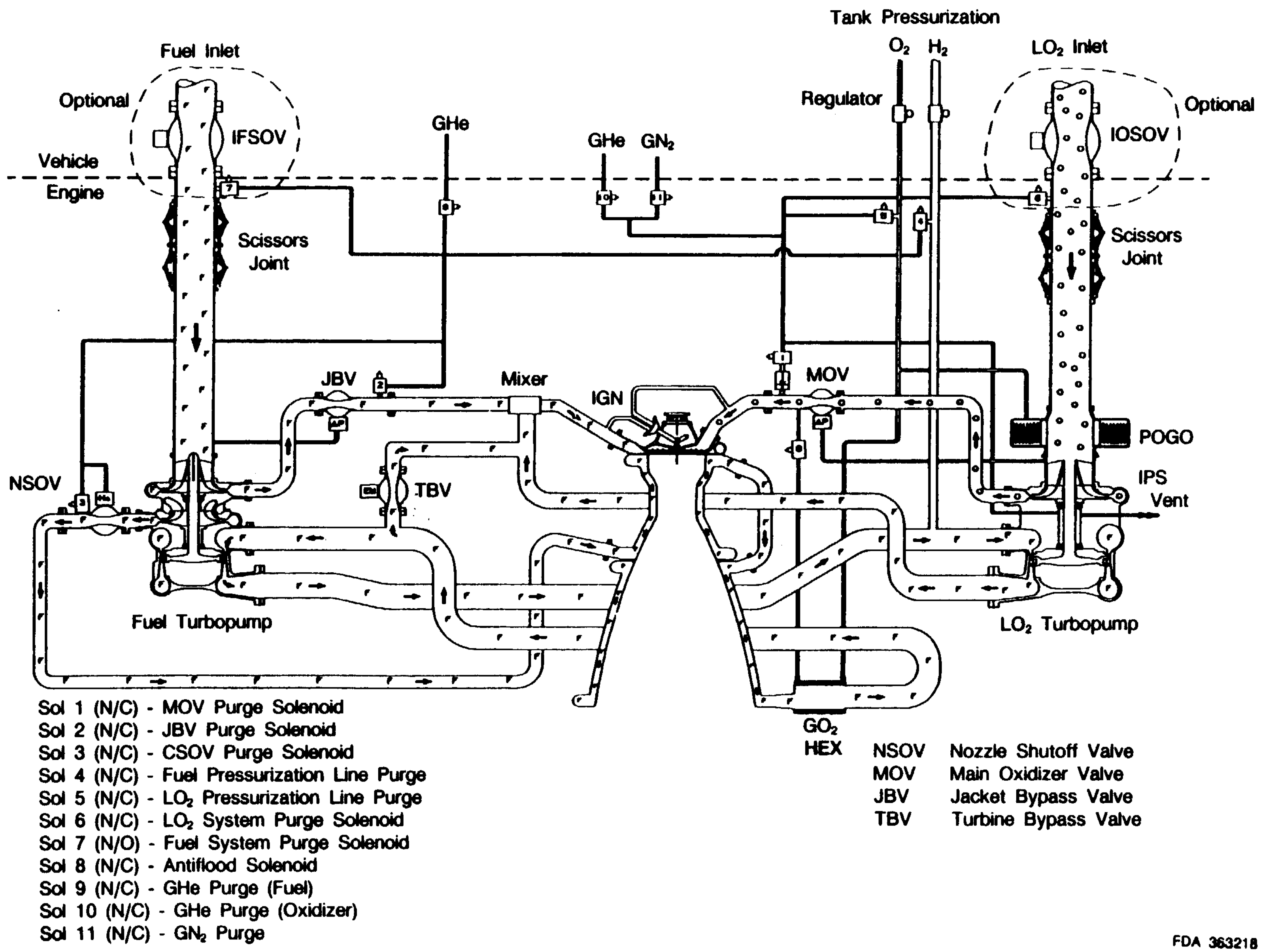


Figure 3.5.2-1. Simplified Flow Schematic for STBE Derivative Split Expander Cycle Engine

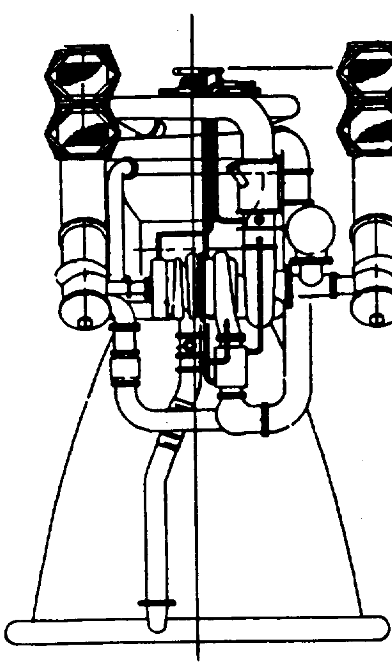
During preconditioning, all of the valves are closed except for the MOV; which is approximately 25 percent open for simultaneous LO_2 injector cooldown. Once the engine is adequately preconditioned, the MOV opens further to completely fill the LO_2 injector prior to ignition. During the process of filling the injector, the NSOV remains closed to prevent cooling of the nozzle/chamber cooling jacket. Once the LO_2 injector is full, the NSOV and the FSOV are opened so the fuel can flow freely to the injector. At this point, the JBV and the TBV remain closed to force all of the available fuel through the turbines. After ignition and upon sufficient power from the turbines, the JBV opens to bypass flow from the pump first-stage discharge to the mixer. Once the desired thrust level is achieved, the TBV opens to control turbine power. At this point, the engine should be at its steady-state conditions.

Engine shutdown is accomplished using a time based scheduling of the propellant valves. First, the TBV is further opened to reduce turbine power and slow the pumps. Then the methane system is shut down by closing the JBV, NSOV and FSOV in that order to purge the fuel system of excess methane. Finally, the oxidizer system is shut down by closing the MOV.

3.6 UNIQUE LO_2/CH_4 SPLIT EXPANDER CYCLE ENGINE

3.6.1 Engine Design Evolution

The STBE LO_2/CH_4 Split Expander Engine Study was initiated during the second quarter of 1988 as a Normal Power Level (NPL) design at 625K lbf sea level thrust. This engine was discussed in FR-19691-3 including flow schematic and cycle description, and is shown in Figure 3.6.1-1.



Propellants	CH_4/LO_2
Mixture Ratio	3.5
Chamber Pressure - psia	877
Thrust - Vacuum	762,900
Sea Level - lb	625,000
Specific Impulse - Vacuum	342.8
Sea Level - sec	280.8
Nozzle Area Ratio	20
Diameter - in.	136
Length - in.	205
Weight - lb	6394

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Figure 3.6.1-1. STBE LO_2/CH_4 Unique Split Expander Engine at Normal Operating Conditions

Further engine study refined the design through the last quarter of 1988. The significant changes from the initial design included the elimination of low-pressure boost pumps and the increased thrust to 750K lbf sea level as the Design Power Level. The engine assembly and major characteristics are shown in Figure 3.6.1-2.

3.6.2 Engine Cycle

The STBE (SE) is a split expander cycle with liquid oxygen and liquid methane as the propellants. This engine operates at a main chamber pressure of 764.5 psia at the normal power level (NPL) of 625K lb and has the capability of running at a design power level of 750K lb. The nozzle area ratio is optimized, for a booster engine application, at 13.5:1 and results in a delivered sea level specific impulse of 281.4 seconds at NPL. Figure 3.6.1-2 presents selected engine characteristics at the normal power level.

3.6.2.1 Flowpath Description

A simplified flow schematic for the STBE (SE) is presented in Figure 3.6.2-1 showing only the major flow paths and components.

Liquid oxygen and methane enter the engine at a NPSH level, supplied by the vehicle, sufficient for the high-speed, high-pressure pumps. No boost pumps are required in these systems.

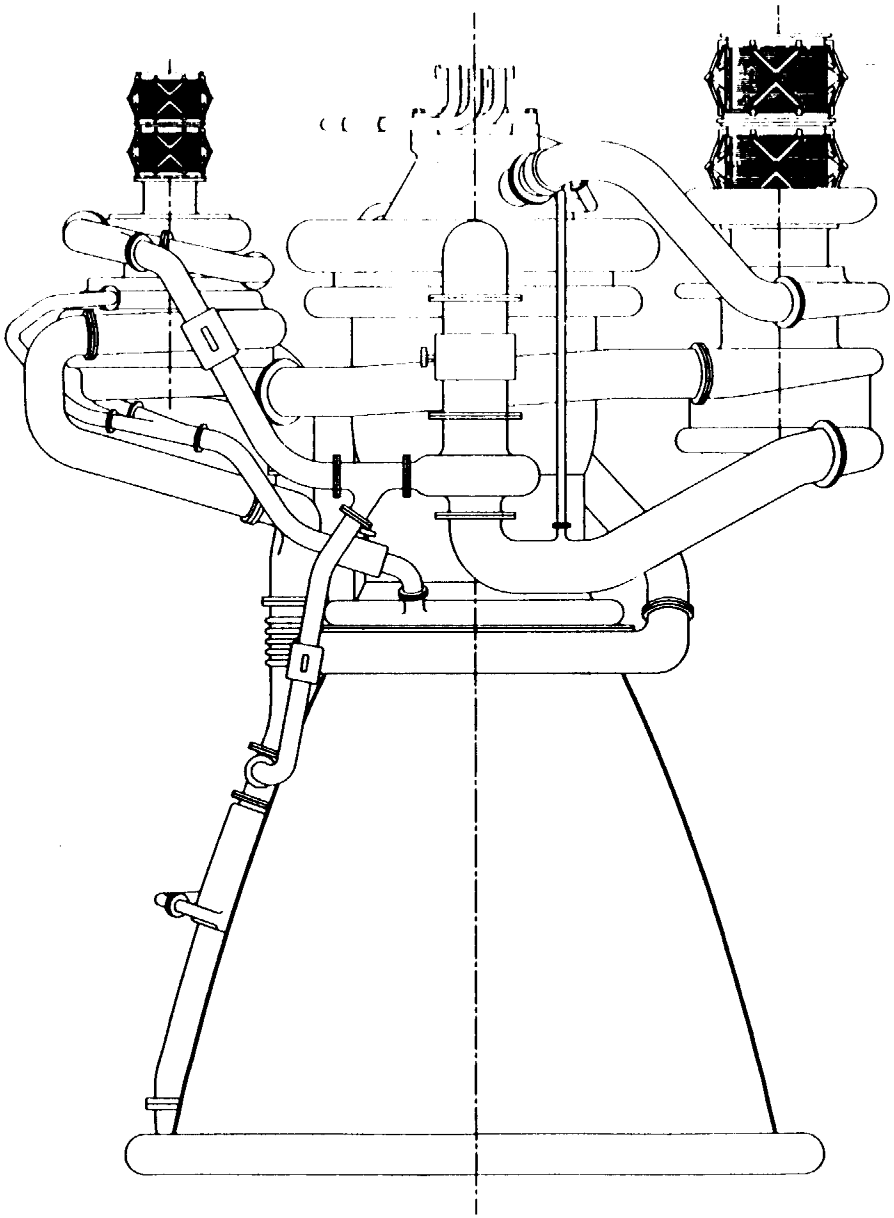
At normal power level, the methane pump operates at 9,689 rpm to provide a first stage pump discharge pressure level of 1098.6 psia. From the first stage pump exit, 44 percent of the fuel is sent through a control valve (JBV) to a mixer downstream of the turbines bypassing the chamber jacket and turbines. The remaining 56 percent of the flow is routed to the second stage of the methane pump. The second-stage pump discharge level is 4072 psia. From the second stage pump exit, the methane is routed through the nozzle shutoff valve into the chamber wall passages where there is counterflow cooling and then through the tubular nozzle wall where there is parallel flow cooling. This heated methane is then used to provide power to drive the propellant pumps. 87.3 percent of the nozzle cooling flow is routed through the turbines. The hot (920 R) methane gas is initially expanded through the methane pump drive turbine and is subsequently routed to a second turbine that powers the oxygen pump. The turbine exhaust is then routed to a mixer, where it combines with the remainder of the methane flow, and is then injected into the main chamber.

At normal power level, the oxidizer operates at 4054 rpm to provide an oxygen pressure level of 978.0 psia. From the pump exit, the oxygen flow is routed through a control valve and injected directly into the main chamber.

3.7 UNIQUE LO_2/CH_4 TAP-OFF CYCLE ENGINE

3.7.1 Engine Cycle

The candidate STBE configuration studied during the Phase A contract is a tap-off cycle with liquid oxygen and liquid methane as propellants. This engine operates at a main chamber pressure of 2400 psia at the rated power level (RPL) of 750,000 pounds thrust. The engine has a fixed nozzle with an area ratio of 35:1 and delivers 305 seconds of sea level specific impulse at RPL. Table 3.7.1-1 presents selected engine characteristics at the rated power level.



	Normal Power Level	Design Power Level
Propellants	LO_2/CH_4	LO_2/CH_4
Mixture Ratio	3.5	3.5
Chamber Pressure	765 psia	895 psia
Thrust - Vacuum	730,389 lb	855,390 lb
- Sea Level	625,000 lb	750,000 lb
Specific Impulse - Vacuum	328.9 sec	329.3 sec
- Sea Level	281.4 sec	288.7 sec
Nozzle Area Ratio	13.5	13.5
Diameter	109 in.	109 in.
Length	170 in.	170 in.
Weight	TBD lb	TBD lb

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Figure 3.6.1-2. STBE LO_2/CH_4 Unique Split Expander Engine at Design Operating Conditions

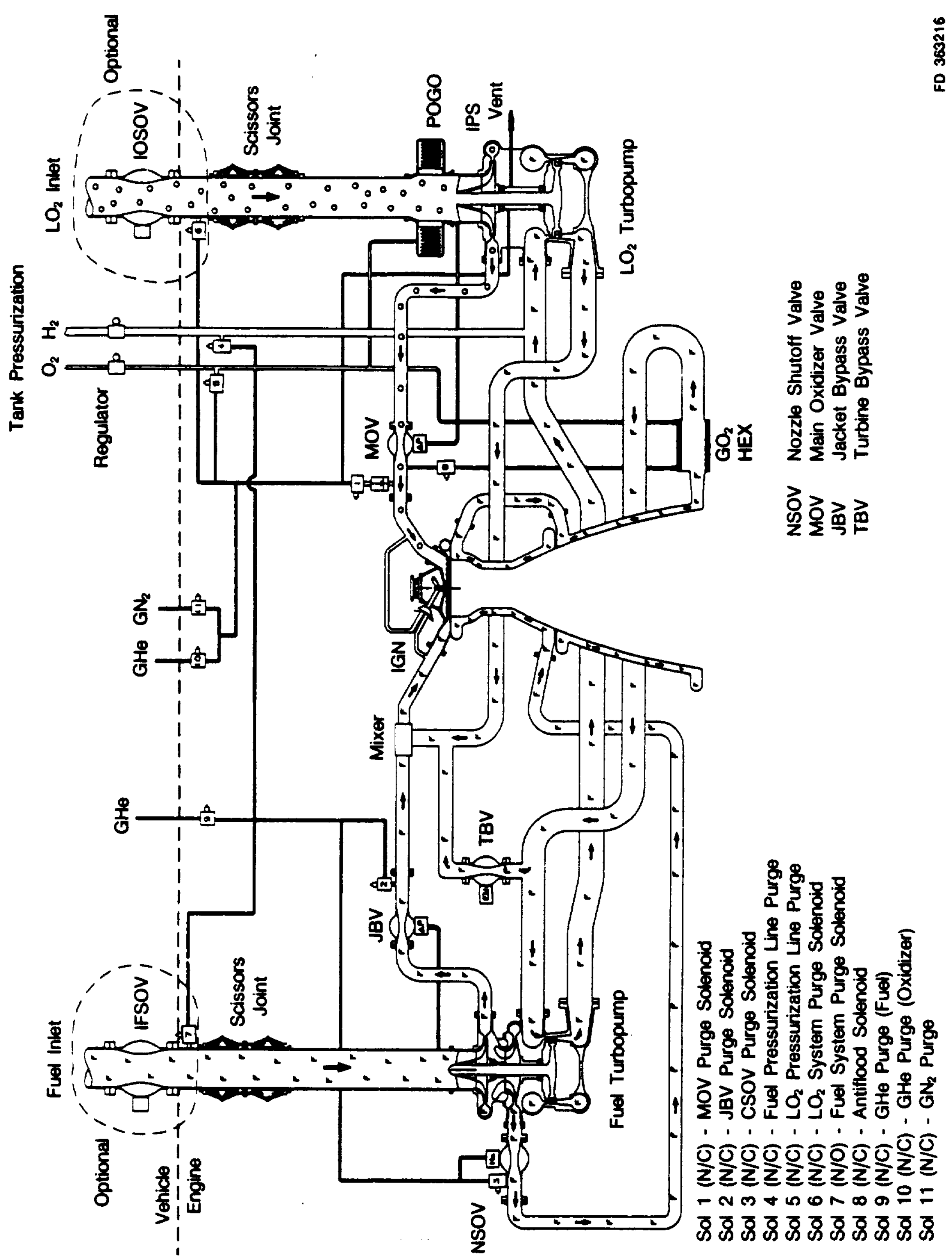

Figure 3.6.2-1. Simplified Flow Schematic for STBE LO_2/CH_4 Unique Split Expander Engine

Table 3.7.1-1. STBE Tap-Off Engine Characteristics — Rated Power Level

Performance	*Tap-Off*
Thrust - lb	750,000
Chamber Pressure - psia	2400
Mixture Ratio	3.0
Specific Impulse (Vac) - sec	342
Area Ratio	35

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3.7.1.1 Flow Path Description

A simplified flow schematic for the STBE tap-off engine is presented in Figure 3.7.1-1 showing the major flow paths and components.

Liquid oxygen enters the engine at a net positive suction head (NPSH) level, supplied by the vehicle, sufficient for the high-speed high-pressure oxidizer pump. Liquid methane enters the engine at a NPSH level, again supplied by the vehicle, sufficient for the high-speed high-pressure methane pump, thus boost pumps are not required for this system.

At the rated power level, the methane pump operates at 16,295 rpm to provide the methane pressure level of 4368 psia required by the cycle. From the pump exit, the methane flows through the fuel shutoff valve where 85.7 percent of it flows to the inlet of the nozzle coolant passages. This methane regeneratively cools the tubular, stainless steel nozzle and milled channel, copper alloy main chamber. From here, the methane flows directly to the injector face. The remaining 12.5 percent of the methane flows through the fuel bypass valve and into the hot gas mixer.

The high-pressure oxidizer pump operates at 6,844 rpm to provide the oxygen pressure level of 3144 psia required by the cycle at the rated power level. From the pump exit, the oxygen flows through the main oxidizer control valve and is injected into the main chamber.

The tap-off provides 1.9 percent of the O/F biased chamber flow to the mixer inlet where cold methane mixes with the hot gases to provide 2293 psia, 1800 R gas to drive the high pressure propellant pumps. This mixed gas then flows through the hot gas valve to the inlet of the methane turbine. The hot gas is initially expanded through the methane turbine and is subsequently routed to a second turbine which powers the oxygen pump. The turbine exhaust gas is then expanded through an area ratio of 5:1 to atmospheric pressure providing additional thrust to the overall engine output.

3.7.1.2 Engine Operation

The engine will be preconditioned using liquid flow from the tanks to soak the turbopumps until they are sufficiently cooled. The inlet valves will be opened, allowing liquid from the tanks to flow down to the turbopumps and letting any vapors percolate back up to the tank to be vented.

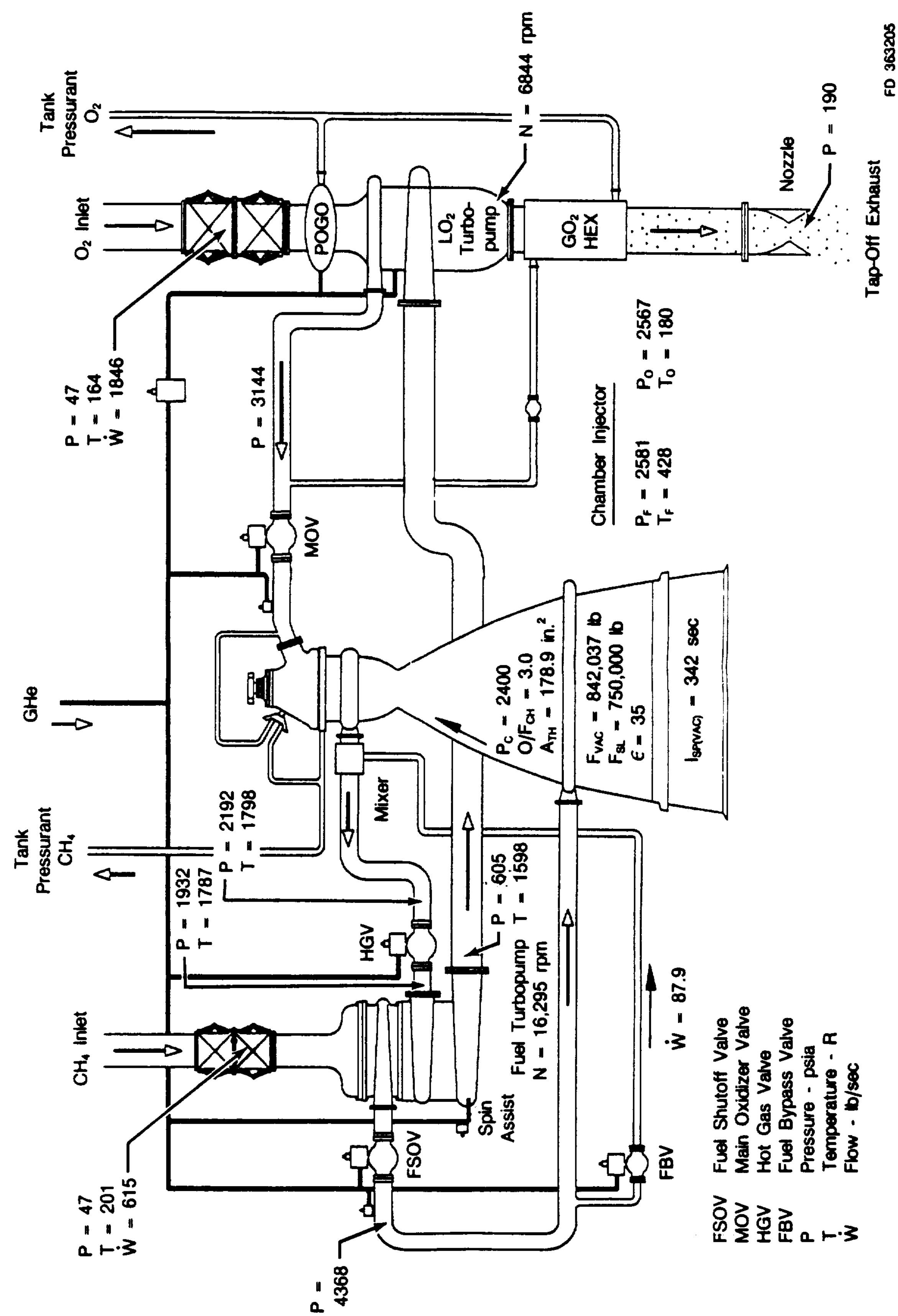

Figure 3.7.1-1. Simplified Flow Schematic for LO_2/CH_4 STBE Tap-Off Cycle Engine at Rated Power Level

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The engine start is a timed sequence process using an oxidizer lead for reliable soft propellant ignition. The oxidizer lead avoids hazardous buildup of unburned fuel in the combustor during the oxygen phase transition from gas to liquid. The transition occurs prior to fuel injection and the fuel is consumed immediately upon injection. Reliability of ignition is enhanced by the LO_2 lead because the transient mixture ratio during propellant filling includes the full excursion of ignitable mixture ratios from greater than 200 to less than one.

With the oxidizer lead sequence, the main chamber LO_2 injector is primed prior to opening the fuel shutoff valve to ensure liquid oxygen flow, eliminating turbine temperature spikes due to oxygen phase change. A helium spin assist is also used to initiate turbopump rotation before the fuel is introduced into the main chamber. During the start and shutdown, a small helium purge is used in the main chamber injector to eliminate the danger of hot gas flow reversals during transient operation. Main chamber ignition will be accomplished with dual electrical spark-excited, oxygen/methane torch igniters.

Main stage engine operation is open-loop controlled. The fuel bypass valve (FBV), the hot gas valve (HGV), and the main oxidizer valve (MOV), shown in Figure 3.7.1-1, are used to set the engine thrust and mixture ratio. Thrust and main chamber mixture ratio are set on the ground by trimming the HGV and MOV, respectively. The turbine inlet temperature is set using the FBV. All valves are operated by hydraulic actuators.

Engine acceleration is accomplished by a time-based scheduling of the valves to the commanded starting level (~ 20 percent power level). The acceleration to full thrust is also accomplished with open-loop valve schedules. Engine shutdown is accomplished using a time-based scheduling of the propellant valves. The HGV is closed first to power down the turbopumps, then the MOV closes, followed by shutting off the methane system.

In addition to a normal operational mode, the engine system is capable of shutdown resulting from detected problems or LO_2 starvation at the end of the burn duration.

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SECTION 4.0
STBE PROGRAMMATIC ANALYSES AND PLANS

Introduction

The following section describes the development plan for the Derivative STBE Gas Generator Engine following the ground rules established by NASA in late 1988 and as summarized in a NASA DDT&E ground rule document dated 20 December 1988. The basic requirement is for a 90-month DDT&E program through Final Flight Certification for an STME engine and an STBE engine derived from the STME.

The objective of the STME DDT&E program is to develop a 580,000-pound vacuum thrust LO_2/LH_2 rocket engine to be used on the core vehicle. The derivative STBE engine is to be a LO_2/CH_4 rocket engine which uses as much hardware common to the STME engine as possible. The resulting derivative STBE has a vacuum thrust of 706.5K pounds and sea level thrust of 500K pounds. Seven derivative STBE engines are to be used on the booster and three engines on the core vehicle (for the purposes of the development plan).

Milestone Dates

The milestone dates as specified by NASA and shown in Table 4.0-1 were used to develop the DDT&E plan.

Table 4.0-1. *STME DDT&E Milestone Dates*

Date	*Milestone*
Jan. 1989	Start Advanced Development Program for gas generator, thrust chamber, turbopumps and engine controls.
June 1989	Start STME Phase B
Oct. 1991	Start Full-Scale Development
Oct. 1993	Component and Subsystem Development Test Facility (CSDTF) available
June 1994	First LO_2/LH_2 engine stand available — 2 positions
Sept. 1994	First LO_2/CH_4 engine stand available — 2 positions
Oct. 1994	Two additional test stands available — 2 positions
Aug. 1995	Critical Design Review
Sept. 1996	MPTA stand available (cluster test)
July 1997	Complete Preliminary Flight Certification, deliver first flight engine set with three spare engines
Jan. 1998	Deliver second flight engine set with three spare engines
Apr. 1998	First flight
Oct. 1998	Second flight
Mar. 1999	Complete Final Flight Certification Tests

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DDT&E Ground Rules

A series of ground rules as specified by NASA, and additional P&W ground rules were used to establish the development plan. These ground rules are shown in Table 4.0-2.

Included in the following sections are: the logic network; the schedules; the test facility requirements; and the Environmental Analysis (DR-10). The Work Breakdown Structure (WBS) and program cost estimates are contained in Volume III of this report.

4.1 LOGIC NETWORK

The logic network shown in Figure 4.1-1 is distributed in time phases, starting with Phase A, Technology and Concept Development, and extending through Production. The items addressed to the appropriate depth for each phase are:

- Chamber/injector demonstration
- Engine design, testing, and production
- Facilities, tooling, and special test equipment
- Launch and flight support.

The Phase A items are described throughout this report and each item is addressed in some detail. Phase A should lead into a Phase A' where more detail will be put into the engine design and analysis. The greater level of detail in Phase A' will allow the various plans to be formulated, along with the very critical safety analyses.

One of the items in Phase B is a demonstration of the combustion efficiency, combustion stability, and heat transfer in tests of a full-scale chamber and injector.

An engine Preliminary Design Review (PDR) will be conducted in Phase B. This design review will be made as a result of the design and analysis that supports engineering layout drawings of the selected concept. At this point, the definition of the engine is sufficiently complete to allow all of the items that were previously labeled preliminary to be finalized. This will also allow the creation of the Design Verification and Substantiation (DVS) requirements for the engine components. The chamber and injector DVS requirements can be used to formulate the test plan for the demonstration chamber and injector.

Completion of the engine layout drawings for PDR allows the planning for the support items to be done. This includes the ground support equipment, tooling, operation, and maintenance planning.

At this point, enough definition of the program has been generated to allow the preparation of a comprehensive Phase C/D proposal.

As the program progresses into Phase C/D, the layout drawings can be turned into detail fabrication drawings. The drawings will be used to fabricate the components and to conduct a comprehensive Critical Design Review (CDR). During fabrication and at fabrication completion, the various component parts and assemblies will be subjected to the DVS tests per the DVS plans that were created during Phase B. The same applies to parts necessary for the engine assembly level, such as flow ducting.

Table 4.0-2. STME/Derivative STBE Development Ground Rules

NASA Groundrules

1. 90-month program through FFC
2. Flight Qualified Engine Life — 15 missions
3. STME engine is to be used for core. Derivative STBE is to be used for the booster stage.
4. 0.99 minimum demonstrated reliability at 90 percent confidence prior to first flight for both engines.
5. Component and engine test conducted by P&W at government owned and operated test facilities at Stennis Space Center. The government will maintain the test facilities down to the interface connections with the test article.
6. The government is to supply the propellants and pressurants at no charge to the contractor.
7. 960 total engine firings through flight testing and final flight certification — applies to the STME. (P&W has established derivative STBE requirement at 488 total engine firings).
8. Two flight tests of booster and core vehicle from ESMC
9. Booster engines are recovered and refurbished following flight test. Core engines are expended.
10. Flight and MPTA engine spares — one spare engine for every three delivered engines.

Additional P&W Ground Rules

1. 488 Derivative STBE engine firings selected for development requirement and to meet reliability requirement of 0.99 at 90 percent confidence on the derivative STBE.
2. STME design, fabrication and testing lead the derivative STBE.
3. Design verification tests on the same or similar STME/Derivative STBE component will be conducted with the higher load set.
4. Conduct verification test with CH_4 on common parts.
5. Hardware design life: 120 firings

   Maximum firing on development hardware: 60 firings.

   Maximum tests between overhauls: 30 firings.
6. Rig mount time (GG and pumps)

with minimal instrumentation:	1 week *
with extensive instrumentation:	2 weeks *
Rig dismount time:	1 week

   * Add one week for main combustion chamber rig.
7. Engine mount time:

with minimal instrumentation:	1 week
with extensive instrumentation:	2 weeks
Engine dismount time:	1 week

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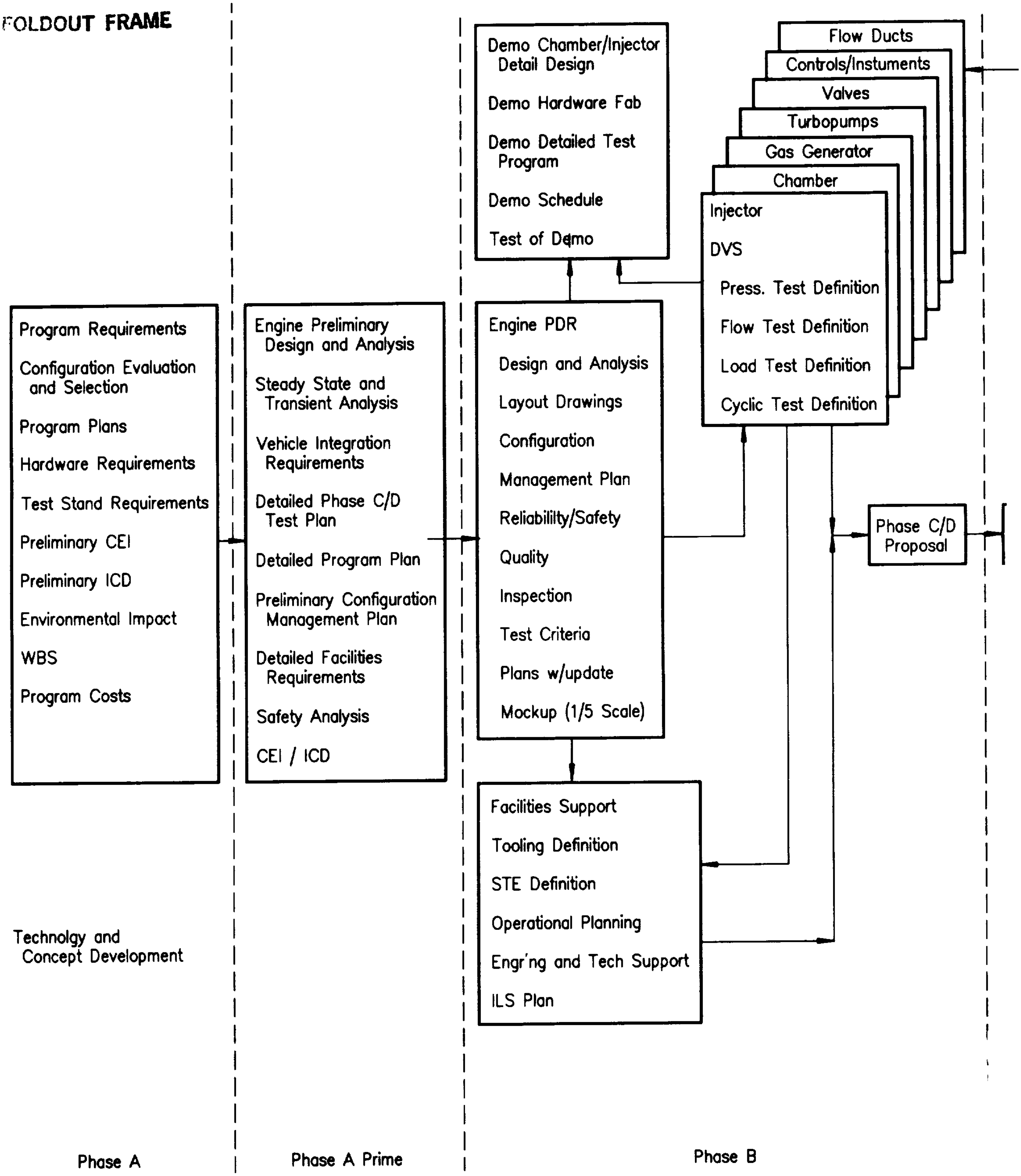

Figure 4.1-1. Logic Network

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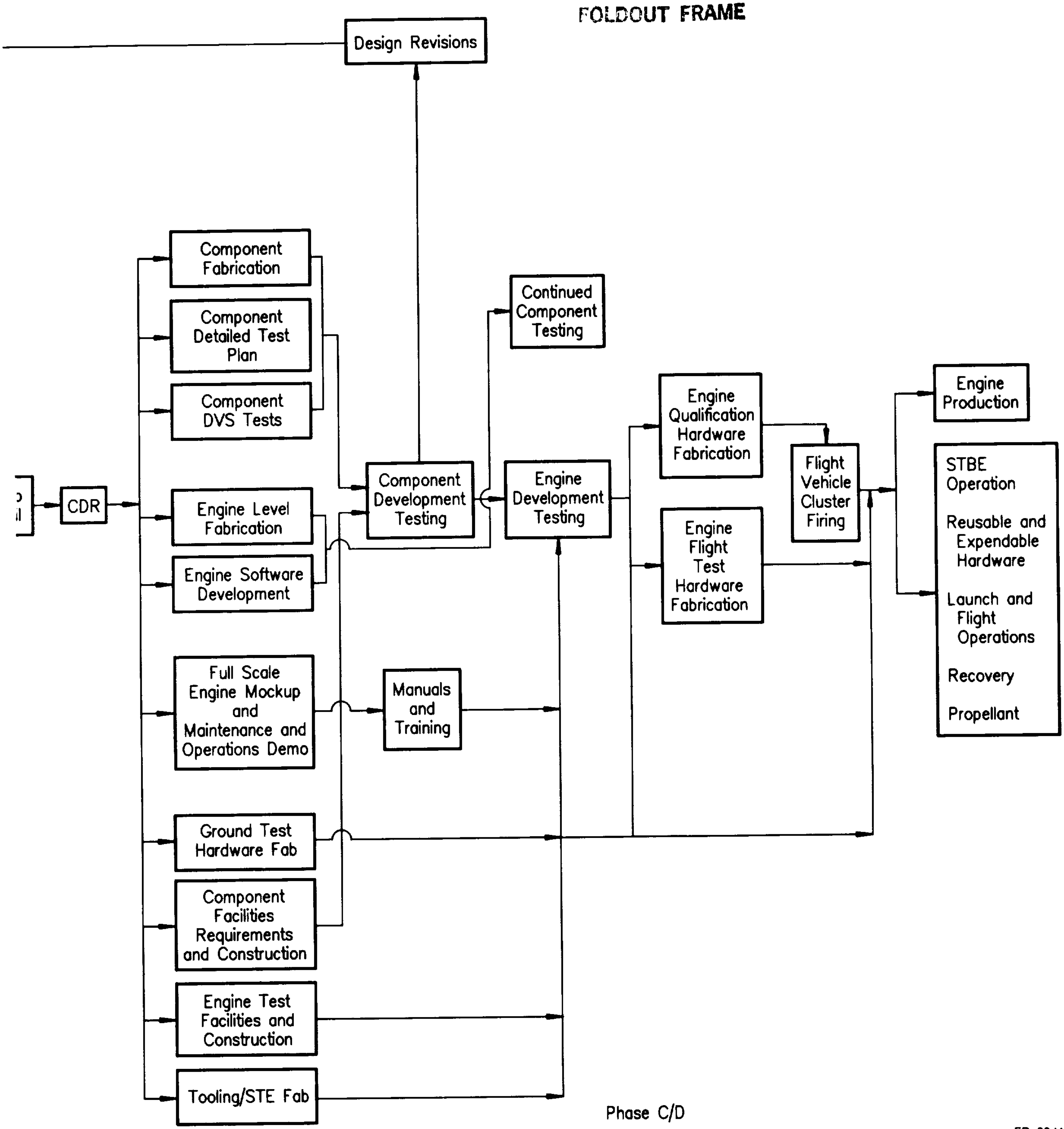

Phase C/D

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4.2 PROJECT SCHEDULES

4.2.1 Major Rig and Engine Tests

The Development Tests scheduled for the DDT&E Program are structured to evaluate and demonstrate all of the functional, durability and performance requirements of the engine. Initially, the component rigs (gas generator, main chamber assembly, LH_2 turbopump and LO_2 turbopump) lead the engine test to ensure that the component has sufficient performance, function and durability to qualify the component for integration into the engine. The rigs will also be used to evaluate part redesigns prior to introduction into a development engine. The rigs will be used in the development program up to the time that engine firings commence for the preliminary flight certification of the engine. At this time sufficient confidence should be demonstrated that the engine is safe to operate and any additional part changes can be evaluated in the engines. The number of component and engine tests for the derivative STBE are less than for the STME due to commonality of the majority of the hardware and also since the STME development program will lead the derivative STBE program. The commonality aspects of the derivative STBE are described in paragraph 3.1.2.

It should be noted that the number of rig tests on the derivative STBE LO_2 pump is limited by test facility capacity to 120 test runs. It is desirable to conduct 300 test runs of this pump since it has little commonality to the STME LO_2 pump and 300 runs are preferred when developing a new turbopump. In contrast, the derivative STBE fuel pump and gas generator are similar to the STME and they require fewer component tests than the similar STME component since the STME component will lead the development program.

Several categories of test series are planned for the development of the STME and derivative STBE engine. The first engine test will follow the first rig test by eight months. The major test categories and test objectives are listed below.

Major Test Series	*Test Objectives*
• Functional Checkout	Leakage Tests Gimballing Capability Controller Checkout Health Monitor Checkout
• Interface	Gimbal Rate Tank Pressurization Propellant Inlet Purge
• Environmental/Structural	Acoustic Signature Engine Vibration Acoustic Loads Starting, Operating and Shutdown Loads Thermal Conditioning Component Stress and Vibration

• Operational Demonstration	Prestart Conditioning Ignition Start/Shutdown Rates, Impulse Throttle Command Response Combustion Stability Engine Pressure, Temperature, Flow Rates Engine Redline Limits POGO
• Performance Demonstration	Engine Calibration Thrust Level Specific Impulse Mixture Ratio Tolerance Performance Repeatability
• Development Testing	General development tests conducted on pre-preliminary Flight Certification Configuration engines to verify engine designs and to eliminate potential engine anomalies
• Mission Testing	Tests conducted on Preliminary Flight Certification engine configurations to demonstrate the reliability requirements of the engines. Firings conducted on these engine are all considered to be accountable firings.
• MPTA (Cluster) Tests	Fire all 10 vehicle engines at one time Verify base heating
• Preliminary Flight Certification Testing (PFC)	Sixty firings conducted on two engines to demonstrate durability and operability requirements of the engine specification.
• Development Flight Tests	Experimental flight test and booster engine recovery
• Final Flight Certification Test (FFC)	Sixty firings conducted on two engines to demonstrate final production engine durability and operability requirements of the engine specification. These tests follow the development flight tests.

To demonstrate reliability of the flight configured engine, all engine tests which contribute to the reliability demonstration of the engine must be conducted on hardware which has the configuration of the preliminary flight certification engines. These firings are termed accountable firings since they contribute to the reliability demonstration of the flight configured engine. To demonstrate the required 0.99 reliability at 90 percent confidence a total of 230 engine firings must be successfully accomplished without failure or malfunction of the engine which would require a premature engine shutdown. Alternatively, one malfunction could occur with a total of 388 firings and still meet the reliability requirement. The STME DDT&E Program has been structured to be able to absorb one unanticipated engine failure requiring engine shutdown during the accountable firing phase of the development program without causing a development schedule impact. The derivative STBE program uses 264 accountable firings to demonstrate reliability requirements. Table 4.2-1 lists the engine tests and identifies the number of firings for each type of test. The total number of STME tests is 960 as specified by NASA, of which 414 are accountable firings that occur prior to first flight. The derivative STBE engine uses 488 total engine firings of which 264 firings are accountable prior to one first flight.

Table 4.2-1. *STME/Derivative STBE Development Tests*

	Total Firings		*Accountable Firings Prior to First Flight*	
Engine Tests	*STME*	*DERIV STBE*	*STME*	*DERIV STBE*
• Functional Checkout	15	10		
• Interface	15	10		
• Environmental/Structural	90	45	30	30
• Operational Demonstration	150	30	30	
• General Development (Pre-PFC Configuration)	230	70		
• Mission Testing (PFC Configuration)	258	90	258	90
• Performance Demonstration	40	15		
• Preliminary Flight Certification (PFC)	60	60	60	60
• MPTA	30	70	30	70
• Flight Test (With Checkout)	12	28	6	14
• Final Flight Certification (FFC)	60	60		
Subtotal	960	488	414	264
Total	1448			

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4.2.2 Development Schedules

A Development Schedule of the STME/Derivative STBE Gas Generator engine is shown in Figure 4.2-1. These schedules show the Advanced Development Program which precedes the start of full-scale development. Major milestones are listed at the top of the first sheet which also shows the major component rig (GG, TCA, Turbopump) tests. The second sheet shows the major engine development tests and the qualification tests.

Four engine test stands (each with two positions) are used for the STME/Derivative STBE Development Program. The Component and Subsystem Development Test Facility (CSDTF) is used for the Component Development Tests. One CSDTF test position is used for the GG, one for the TCA, two positions for the LH_2 turbopump, and two positions for the LO_2 pump. The maximum test rate was assumed to be eight firings (runs) per month for each position in the CSDTF and 10 engine firings per month for each engine test position.

As component fabrication is completed, component testing will be conducted. Information obtained during the component tests will allow design revisions necessary to optimize the hardware design. This provides a feedback loop into the DVS planning activity.

As a result and as a part of the detail design effort, the 1/5-scale mockup can be replaced with a full-scale mockup. This mockup greatly facilitates the design of the external flow ducting and allows a demonstration of engine maintenance and operation. The mockup and demonstrations done with the engine will allow the creation of manuals and training material.

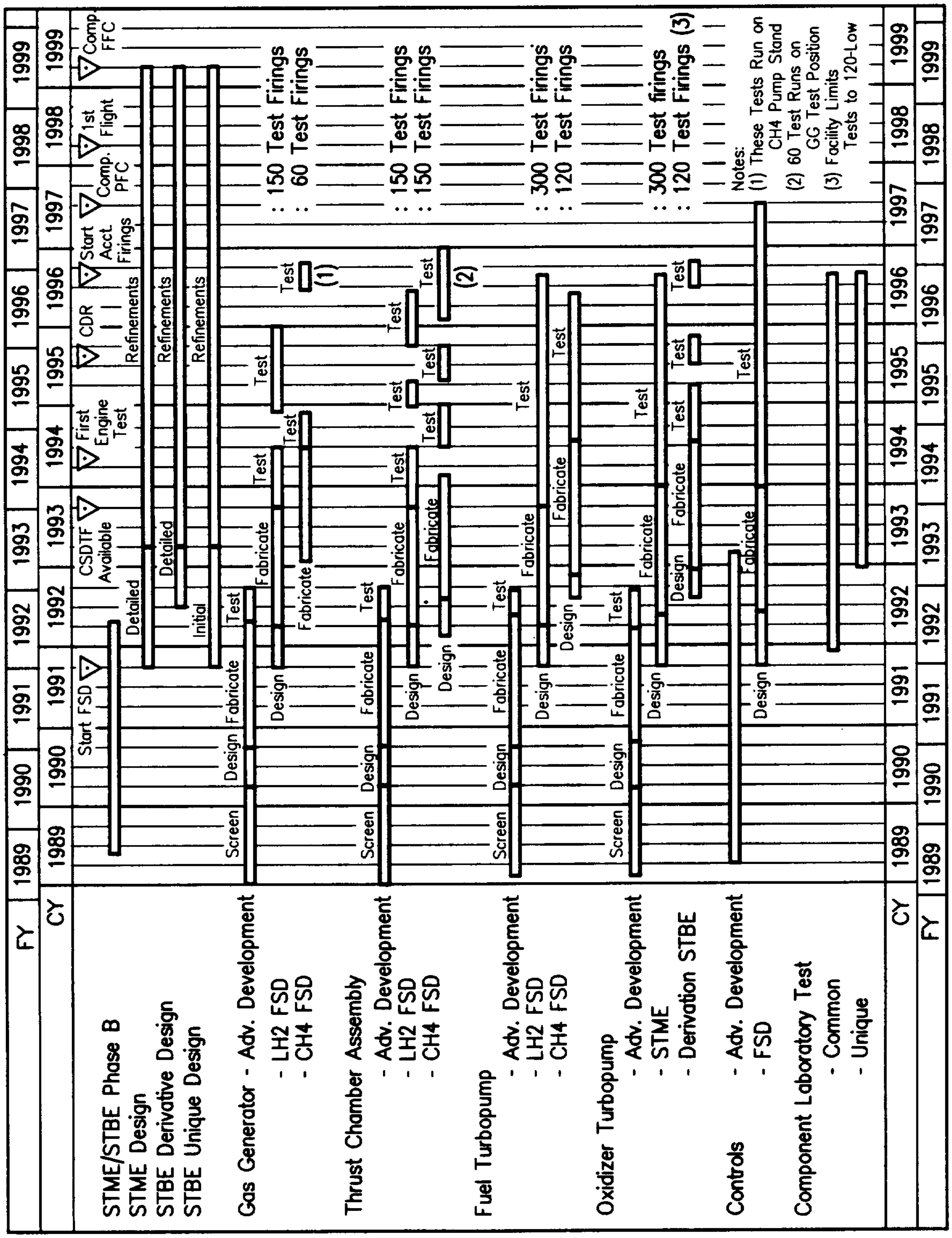

Figure 4.2-1. STME/Derivative STBE Development Schedule (Sheet 1 of 2)

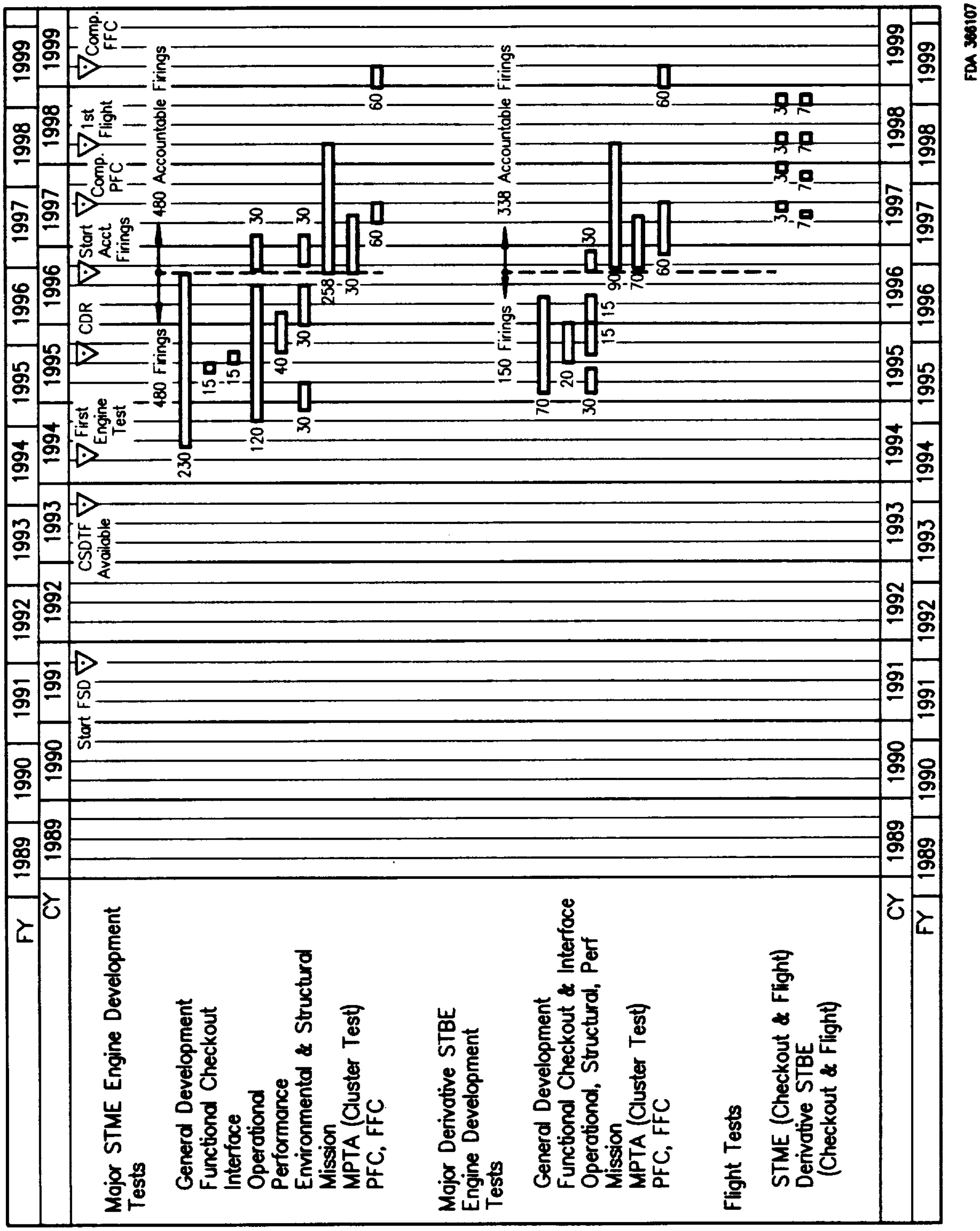

Figure 4.2-1. STME/Derivative STBE Development Schedule (Sheet 2 of 2)

The ground test hardware, tooling, and special test equipment necessary for engine testing and STBE operation will be fabricated during this phase.

Early in Phase C/D, the engine contractor must participate in the engine test facilities requirements and follow the test stand fabrication. When the initial tests of the component hardware are completed, the components can be assembled together to conduct engine development tests. Component tests will continue in parallel to accumulate confidence that test time-related malfunctions have been found and corrected.

As engine test time is accumulated and design iterations diminish, the design can be frozen and hardware for engine qualification and flight test can be fabricated.

One of the major elements during Phase C/D will be a firing of a cluster of engines with a stackup of vehicle tankage, etc.

The program then progresses into engine production and engine operation activities.

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SECTION 5.0
STME/DERIVATIVE STBE PROGRAM COST ESTIMATES

Program cost estimates were made for the gas generator STME/Derivative STBE program as part of Task V (SOW Task 5.4) using the currently approved Space Transportation Engine Work Breakdown Structure (WBS). All engine related design and development, operational production, operations and product improvement and support program cost elements were included in the estimates. Brief summaries of the estimated costs are presented in this section. The details of these cost estimates are contained in the Program Cost Estimates Document (DR6) which is Volume III of this report. Volume III also contains the Work Breakdown Structure and WBS Dictionary (DR5) used for the cost estimates.

All costs generated in this study are Rough Order Magnitude (ROM) engineering estimates. The costs estimated are a function of the ground rules assumed for the program. The costs should not be construed as contractual commitments and should be used for Life Cycle Cost (LCC) evaluation and program planning purposes only.

The approved Work Breakdown Structure used for these estimates is shown in Table 5-1 and Figures 5-1 and 5-2. Table 5-1 illustrates how the engine WBS fits into the overall Advanced Launch System WBS. Figure 5-1 shows the WBS functional elements and hardware breakouts used for the engine Design and Development Phase (Phase C/D) cost estimates while Figure 5-2 shows the WBS elements used for the Operations Phase (Phase E) cost estimates.

Table 5-1. *Advanced Launch Vehicle System WBS*

WBS No.	*Space System Work Breakdown Structure Elements*
1.0	Advanced Launch System
1.1	System — Integration, Assembly and Test
1.2	Launch Vehicle System
1.2.1	Launch Vehicle System — Integration, Assembly and Test
1.2.N	Vehicle Stage (N=2 Booster, N=3 Core)
1.2.N.7	Liquid Fuel System
1.2.N.7.2	Main Engines
1.2.N.7.2-1	Main Engines — Design and Development
1.2.N.7.2-2	Main Engines — Non-Recurring Operational Production
1.2.N.7.2-3	Main Engines — Recurring Operational Production
1.2.N.7.2-4	Main Engines — Operations
1.2.N.7.2-X	Main Engines — Product Improvement and Support Program

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One ALS scenario (Scenario 2) designated by NASA for the methane booster was evaluated for the STME/Derivative STBE cost estimates. The Scenario 2 vehicle, which is shown in Figure 5-3, consists of a hydrogen/oxygen core stage powered by three reusable STME's, and a methane/oxygen booster stage powered by seven reusable Derivative STBE's. Nominal, maximum and minimum flight schedules and production engine quantities were evaluated for this scenario. The STME used on the core stage is the baseline STME with the nozzle skirt, defined in the Space Transportation Main Engine Configuration Study (See FR-19830-2). The Derivative STBE is the final methane derivative configuration of the STME which has 72% costs commonality with the STME. General ground rules and assumptions used for the cost evaluations are summarized in Table 5-2. The number of missions and quantities of engines assumed for each of the three scenario case are summarized in Table 5-3. Figure 5-4 shows the flight schedules used for each case.

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	WBS No. 1.2.N.7.2-	Prog Mgmt		SE & I		Eng'g D&D	Engine Test					Flight Test Hardware		MPTA Test Hardware		Facilities			Software Engineering	GSE		Tooling	STE	Operations & Support			
		Contract Data	Other	SRM & QA	Other		Engine Test Hardware	Engine Test Operations & Support	Component Test Hardware	Component Test Operations & Support	Propellants	Mfg.	Acceptance	Mfg.	Acceptance	Production	Launch	Test		Common	Peculiar			Launch Operations Support	Flight Operations Support	Recovery Operations Support	Refurbishment Operations Support
Integrated System	1.13	O	O	O	O	O	O	O	O	O		O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O
System Hardware	1.13.1																										
Turbomachinery	1.13.1.1								O	O		O		O													
LPOTP	1.13.1.1.1											O		O													
LPFTP	1.13.1.1.2											O		O													
HPOTP	1.13.1.1.3											O		O													
HPFTP	1.13.1.1.4											O		O													
Combustion Devices	1.13.1.2								O	O		O		O													
Main Injector	1.13.1.2.1											O		O													
Thrust Chamber	1.13.1.2.2											O		O													
Nozzle	1.13.1.2.3											O		O													
Nozzle Skirt	1.13.1.2.4											O		O													
Gas Generator	1.13.1.2.5											O		O													
Igniters	1.13.1.2.6											O		O													
Controls	1.13.1.3								O	O		O		O													
Controllers/Monitors	1.13.1.3.1											O		O													
Software	1.13.1.3.2											O		O					O								
Sensors	1.13.1.3.3											O		O													
Valves	1.13.1.3.4											O		O													
Actuators	1.13.1.3.5											O		O													
Interconnects	1.13.1.3.6											O		O													
Propellant Feed	1.13.1.4											O		O													
Cold Ducts	1.13.1.4.1											O		O													
Hot Ducts	1.13.1.4.2											O		O													
Miscellaneous	1.13.1.4.3											O		O													
Support Devices	1.13.1.5											O		O													
Gimbal System	1.13.1.5.1											O		O													
Tank Repress. Sys	1.13.1.5.2											O		O													
Power Topoff	1.13.1.5.3											O		O													
Start System	1.13.1.5.4											O		O													
POGO System	1.13.1.5.5											O		O													
Integ., Assy & Test	1.13.2											O		O					O								

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Figure 5-1. STME/Derivative STBE Work Breakdown Structure — Design and Development Phase

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	WBS No. 1.2.N.7.2-	WBS No. 1.2.N.7.2-2 Non-Recurring Operational Production											WBS No. 1.2.N.7.2-3 Recurring Oper. Prod.							WBS No. 1.2.N.7.2-4 Operations										
		Prog Mgmt	SE & I		Facilities			GSE		Tooling	STE	Initial Spares	Program Mgmt	SE & I		Flt Hdw Manuf		Tooling Maint	Facilities Maint	Program Mgmt	SE & I		Facilities Maint		Operations & Support					Training
			SRM & QA	Other	Production	Launch	Test	Common	Peculiar					SRM & QA	Other	Mfg.	Acceptance				SRM & QA	Other	Launch	Test	Launch Ops.	Flight Ops.	Spares Rep.	Recovery Ops.	Refurb Ops.	
Integrated System	13	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O
System Hardware	13.1				O	O	O																							
Turbomachinery	13.1.1											O				O														
LPOTP	13.1.1.1															O														
LPFTP	13.1.1.2															O														
HPOTP	13.1.1.3															O														
HPFTP	13.1.1.4															O														
Combustion Devices	13.1.2											O				O														
Main Injector	13.1.2.1															O														
Thrust Chamber	13.1.2.2															O														
Nozzle	13.1.2.3															O														
Nozzle Skirt	13.1.2.4															O														
Gas Generator	13.1.2.5															O														
Igniters	13.1.2.6															O														
Controls	13.1.3											O				O														
Controllers/Monitors	13.1.3.1															O														
Software	13.1.3.2															O														
Sensors	13.1.3.3															O														
Valves	13.1.3.4															O														
Actuators	13.1.3.5															O														
Interconnects	13.1.3.6															O														
Main Propellant Feed	13.1.4											O				O														
Cold Ducts	13.1.4.1															O														
Hot Ducts	13.1.4.2															O														
Miscellaneous	13.1.4.3															O														
Support Devices	13.1.5											O				O														
Gimbal System	13.1.5.1															O														
Tank Repress. Sys	13.1.5.2															O														
Power Topoff	13.1.5.3															O														
Start System	13.1.5.4															O														
POGO System	13.1.5.5															O														
Integ., Assy & Test	13.2															O														

Figure 5-2. STME/Derivative STBE Work Breakdown Structure — Operations Phase

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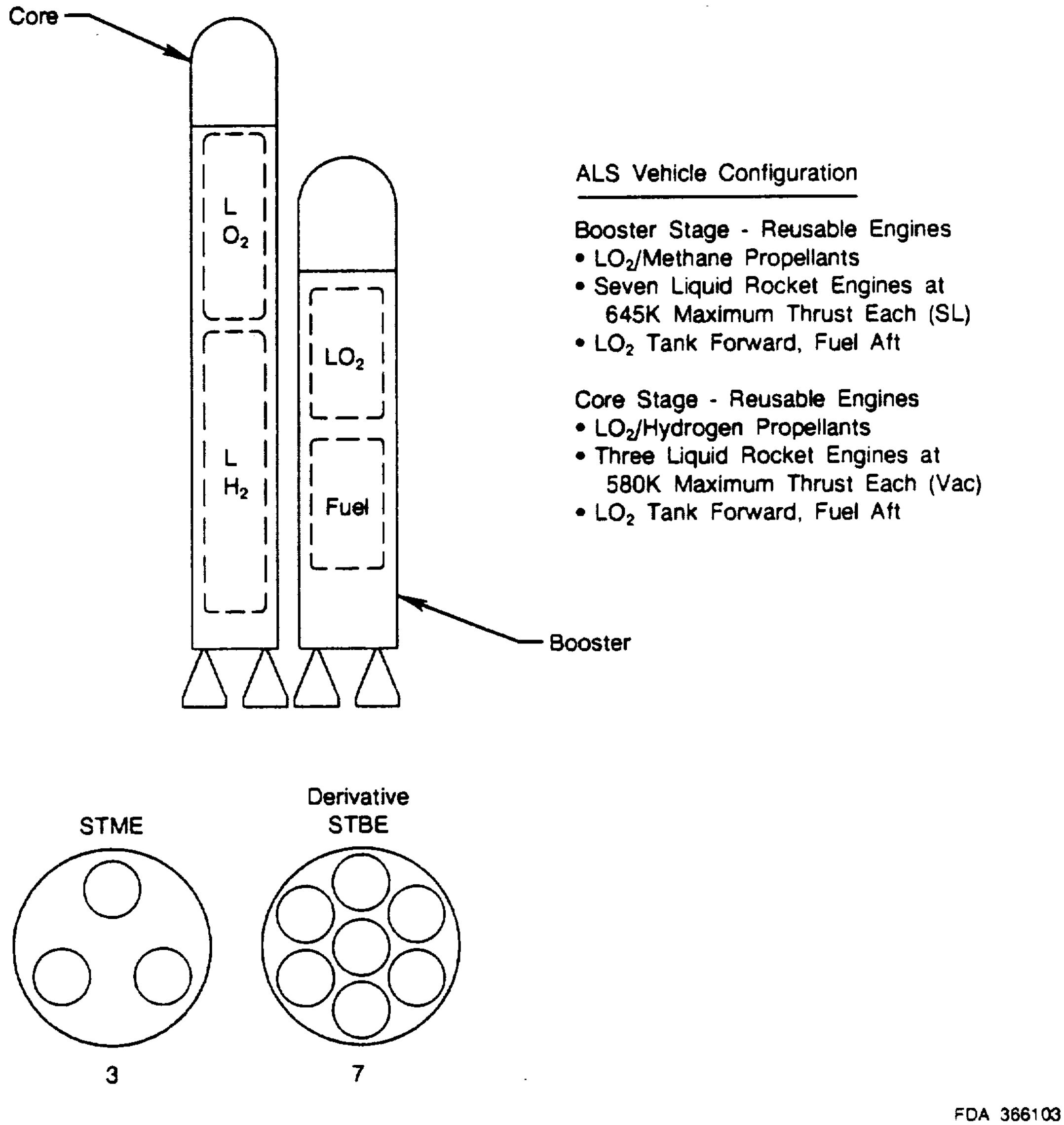

Figure 5-3. ALS Vehicle Configuration

Table 5-2. STME/Derivative STBE Cost Ground Rules and Assumptions

Dollars	Constant FY87
Fee and Management Reserve	Not Included
Engine Test Facilities	Gov't. Provided (Not Included)
Propellants	Gov't. Provided (Not Included)
Engine Configuration	
STME (Core)	580K Vac H_2/O_2 Generator Engine with 62 AR Nozzle Skirt
Derivative STBE (Booster)	645K SL CH_4/O_2 Gas Generator Engine with 28 AR Nozzle Derived from STME
Number Engines/Stage	7 — Booster; 3 — Core
Development Period	7.5 years
Operational Production Period	Varies for each case; See Table 4-2
Operations Period	25 Years
Product Improvement and Support Period	7 Years
Number Launch Sites	1 — ESMC
Production Engine Assembly/Acceptance Test	At SSC
Engine Recovery (Scenario 1)	Booster Engine Ocean Recovered Subjected to Salt Air Only; Core Engine Expended or Land Recovered
Operations Maintenance Levels	Engine/Component Removal and Replacement at ESMC; Component Refurbishment at SSC Depot

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Table 5-3. STME/Derivative STBE Program Cost Scenarios

	Scenario 2					
	Core Stage			*Booster Stage*		
	Nominal	*Maximum*	*Minimum*	*Nominal*	*Maximum*	*Minimum*
Total Number of Missions	300	625	250	300	625	250
Maximum Number of Missions/Year	14	33	12	14	33	12
Total Number of Operational Production Engines	175	350	100	425	850	275
Maximum Number of Production Engines/Year	30	30	30	70	70	70
Average Number of Reuses/Engine	5	5	7	5	5	6
Operational Production Period, Yrs	24	23	9	24	23	12

Note: Scenarios 1 and 3 address STME and are included in FR-19830-2.

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Total program cost estimates for each STME/Derivative STBE Scenario 2 case are summarized in Table 5-4. The nominal STME/Derivative STBE flight case which consists of 300 missions over a 25-year operational period results in a total program cost of approximately \$6.7 billion. This cost is approximately 16 percent higher than the comparable Baseline STME case (See FR-19830-2) which uses reusable hydrogen/oxygen STME's on both stages. The highest cost STME/Derivative STBE case (maximum flight schedule with 625 missions) has a program cost less than \$10 billion (\$9743M).

Design and Development program costs are summarized in Table 5-5 while Operational Production program costs for each case are summarized in Table 5-6. Operations costs for each case are presented in Table 5-7.

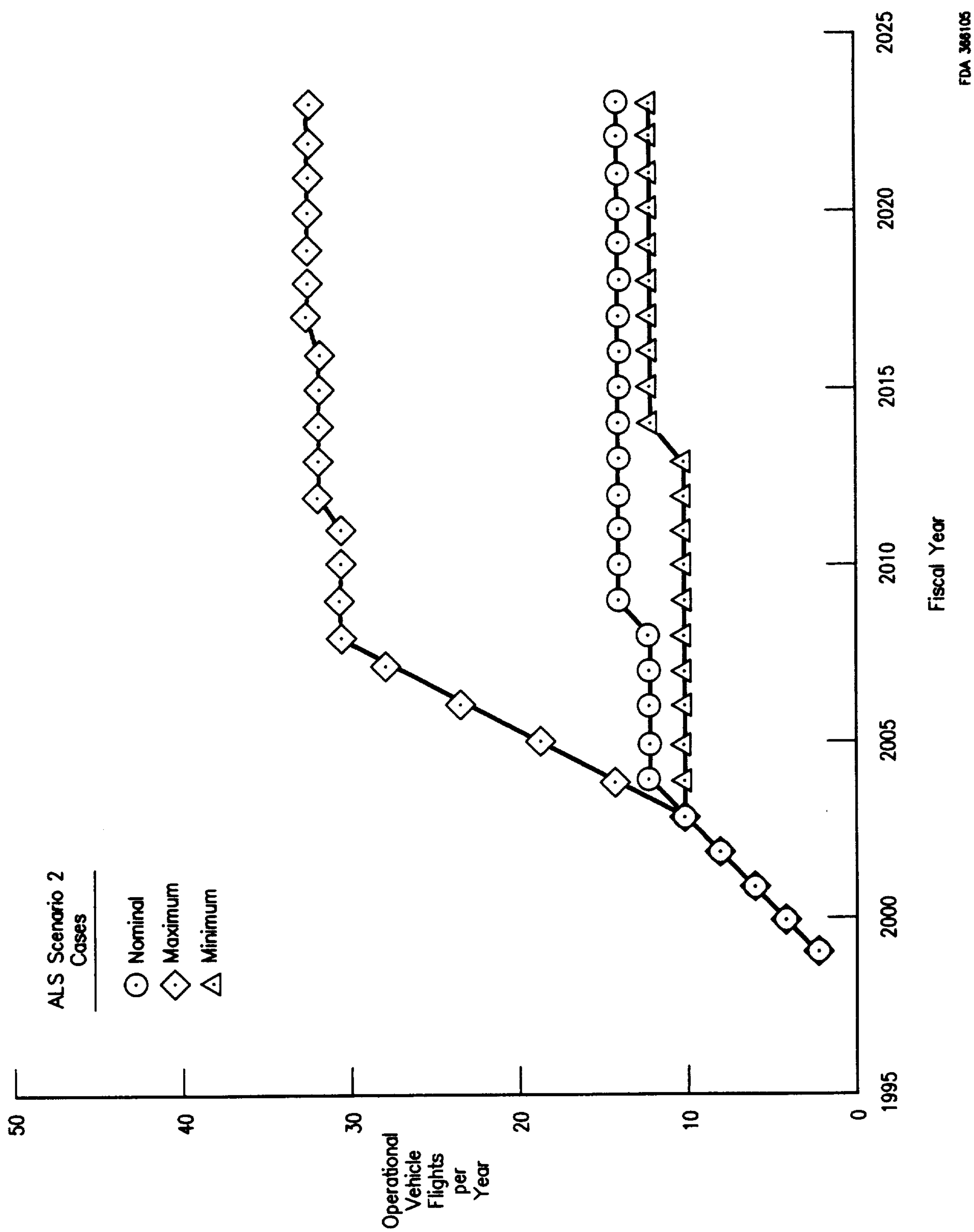

Figure 5-4. STME/Derivative STBE (Scenario 2) Mission Flight Rates

Table 5-4. *STME/Derivative STBE Program Cost Summary*

	Scenario 2 Mission Schedule		
	Nominal	*Maximum*	*Minimum*
Design and Development	$1841.1 M	$1841.1 M	$1841.1 M
Non-Recurring Operational Production	366.4	694.9	352.2
Core Engines	120.0	232.8	112.9
Booster Engines	246.4	462.1	239.3
Recurring Operational Production	3226.3	5728.7	2054.5
Core Engines	1064.1	1890.0	621.7
Booster Engines	2162.2	3838.7	1432.8
Operations	479.5	739.2	437.0
Core Engines	140.1	214.9	128.0
Booster Engines	339.4	524.3	309.0
Product Improvement and Support Program	739.1	739.1	739.1
Total Program Cost	$6652.4 M	$9743.0 M	$5423.9 M

Note: All costs in millions of constant FY87 dollars.

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Table 5-5. *Gas Generator STME/Derivative STBE Program — Design and Development Program Cost Summary*

	STME Portion	*STBE Portion*	*Total*
Program Management	$66M	$13M	$79M
System Engineering and Integration	42	24	66
Engine Design and Development	171	63	234
Engine Test			
Test Hardware	352	184	536
Test Operations and Support	254	110	364
Flight Test Hardware	73	147	220
MPTA Test Hardware	37	70	107
Facilities			
Production	8	0	8
Launch	4	0	4
Test	22	2	24
Software Engineering	12	3	15
GSE	19	9	28
Tooling	68	10	78
Special Test Equipment (STE)	25	5	30
Operations and Support	30	18	48
Total DDT&E Program Cost	$1,183M	$658M	$1,841M

Note: All costs in millions of FY87 dollars.

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Annual funding requirements for the total STME/Derivative STBE program are shown in Figure 5-5 for each of the three cases. Design and Development funding schedules are shown in Figure 5-6 while Operational Production schedules are shown in Figure 5-7. Operations and Product Improvement and Support Program funding requirements are shown in Figures 5-8 and 5-9 respectively.

Table 5-6. *Operational Production Cost Summary STME/Derivative. STBE Program*

	Scenario 2		
	Nominal	*Maximum*	*Minimum*
Non-Recurring Operational Production			
Program Management	$ 3.7	$ 3.7	$ 3.7
System Engineering and Integration	16.1	16.1	16.1
Facilities	0	0	0
Ground Support Equipment	33.0	77.0	33.0
Tooling	48.0	48.0	48.0
Special Test Equipment	0	0	0
Initial Spares	265.7	550.4	251.5
Total Non-Recurring Production Cost	$366.5	$695.2	$352.3
Recurring Operational Production			
Program Management	$ 13.9*	24.7*	8.9*
System Engineering and Integration	111.3	197.5	70.8
Flight Hardware Manufacturing	3,101.1	5,506.8	1,974.8
Tooling Maintenance	0 *	0 *	0 *
Facilities Maintenance	0 *	0 *	0 *
Total Recurring Production Cost	$3,226.3	$5,729.0	$2,054.5
Total Operational Production Cost	$3,592.8	$6,424.2	$2,406.8

* Some recurring program management functions and tooling maintenance and facilities maintenance included in flight hardware manufacturing markups.
Note: All costs in millions of FY87 dollars.

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Table 5-7. *STME/Derivative STBE Program Operations Cost Summary*

	Scenario 2		
	Nominal	*Maximum*	*Minimum*
Program Management	$ 26.5	$ 28.6	$ 26.3
System Engineering and Integration	103.7	112.4	103.1
Facilities Maintenance	0	0	0
Operations and Support			
Launch Operations	15.0	27.0	12.9
Flight Operations	43.2	46.9	43.0
Spares Replenishment	76.9	138.5	66.4
Recovery Operations	12.9	23.2	11.1
Refurbishment Operations	195.3	351.9	168.8
Training	6.0	10.8	5.2
Total Operations Cost	$479.5	$739.3	$436.8

Note: All costs in millions of FY87 dollars.

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Theoretical First Unit recurring production costs for the STME and Derivative STBE are presented in Table 5-8. This table also shows the amount of cost commonality for each component. Operations costs at the engine unit level are presented in Table 5-9. The costs in these tables are the individual engine unit costs used to derive the program cost estimates.

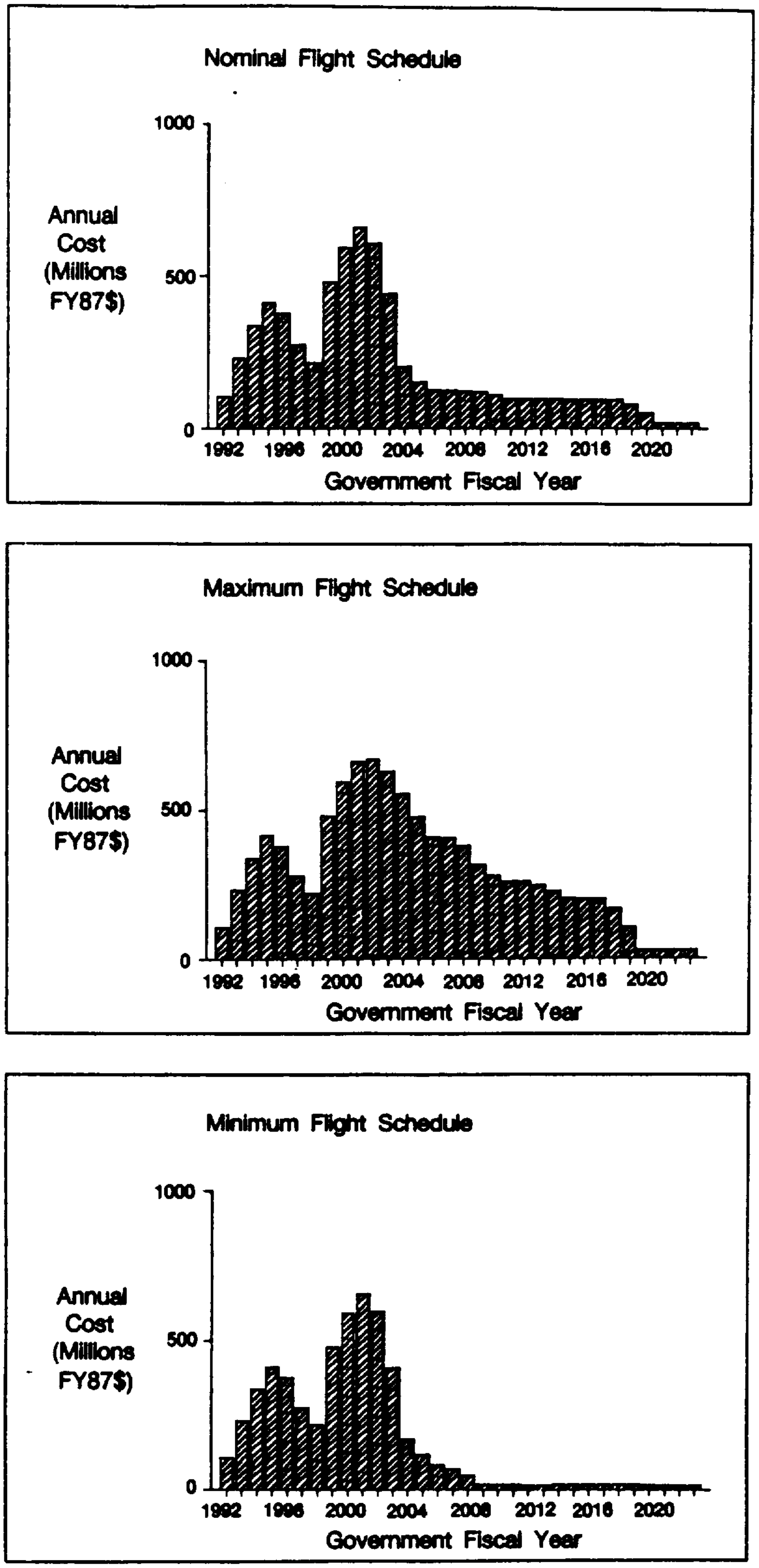

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Figure 5-5. STME/Derivative STBE (Scenario 2) Total Program Costs by Year

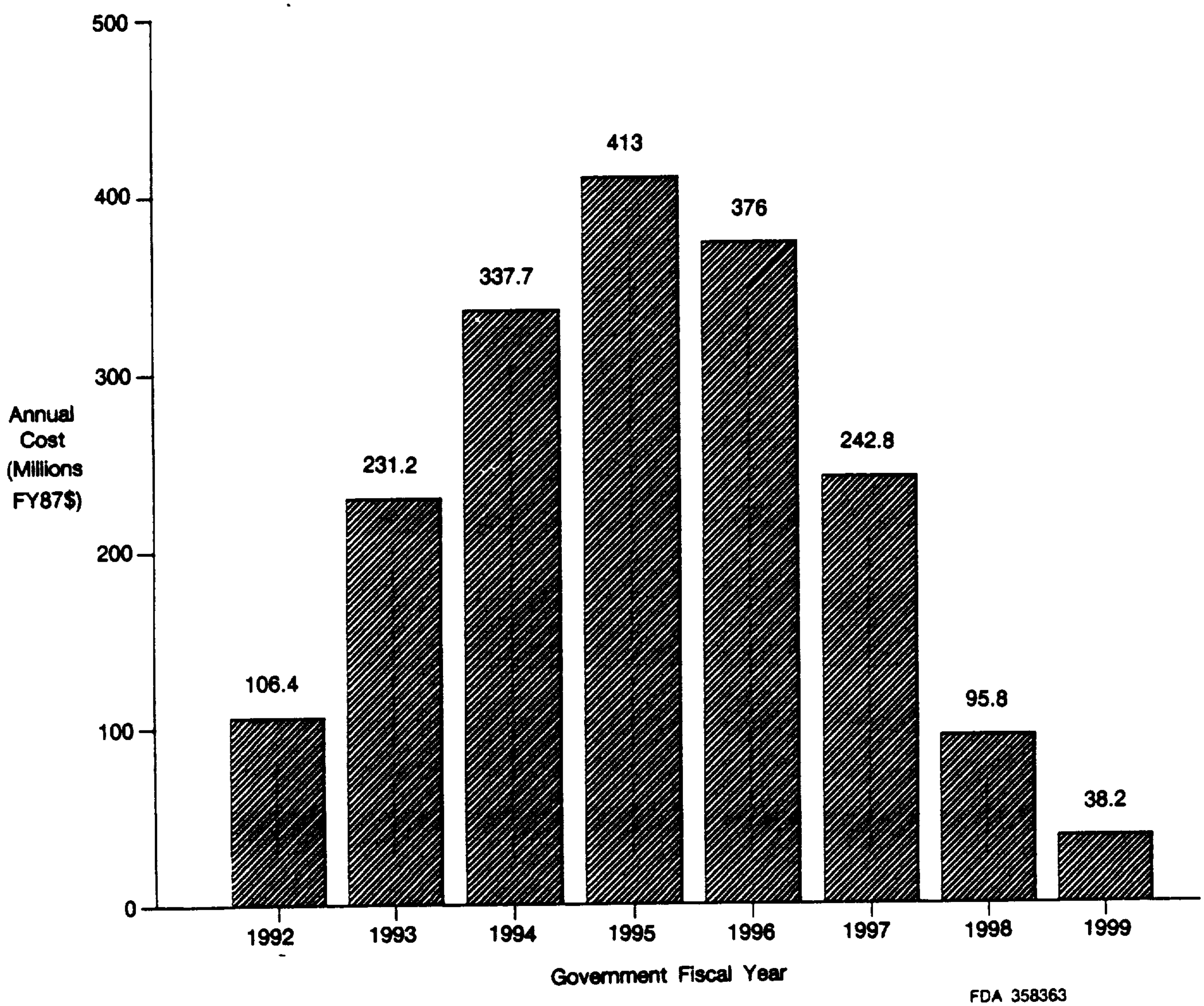

Figure 5-6. STME/Derivative STBE (Scenario 2) Design and Development Phase Costs by Year

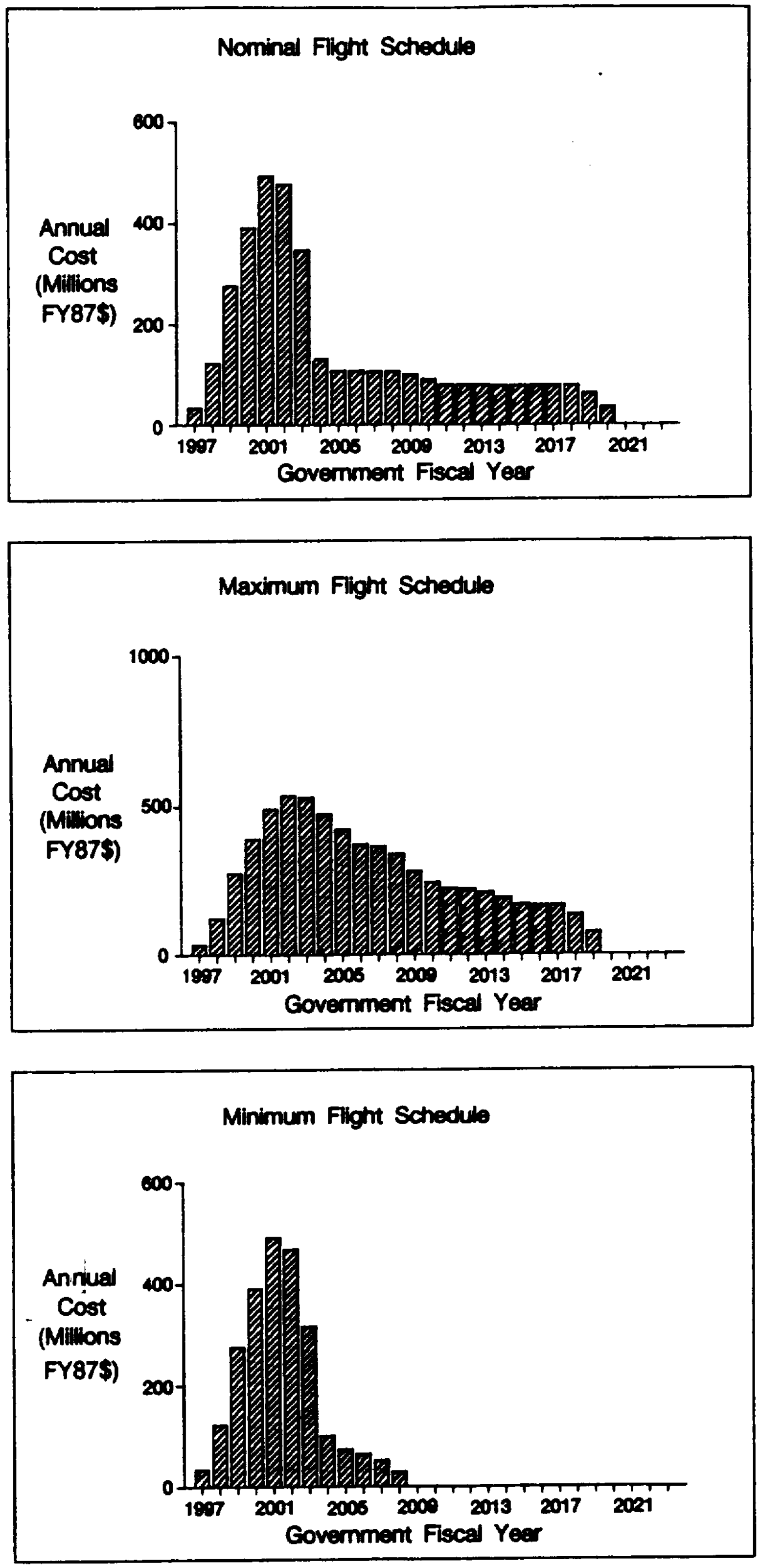

Figure 5-7. STME/Derivative STBE (Scenario 2) Operational Production Costs by Year

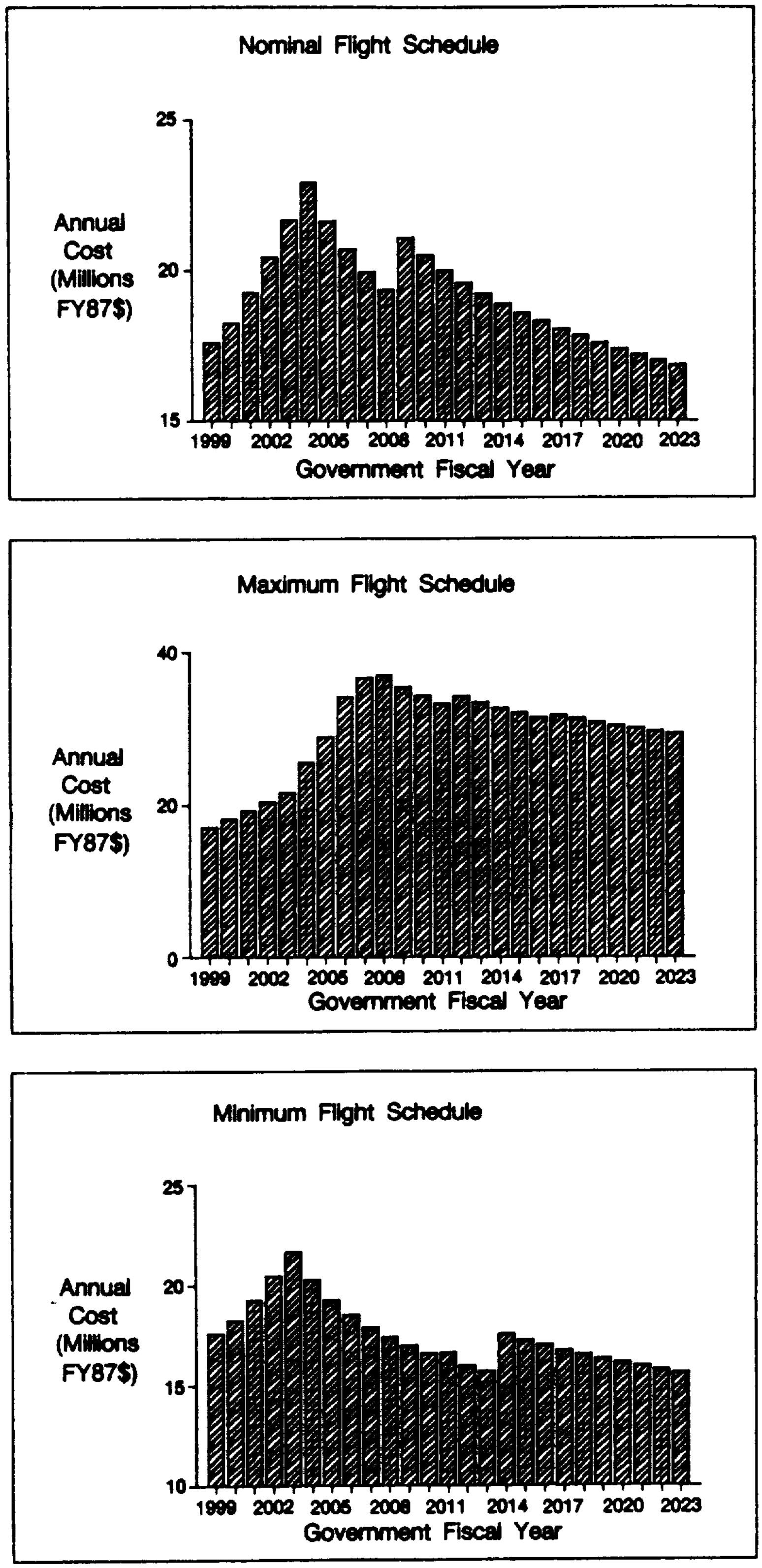

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Figure 5-8. STME/Derivative STBE (Scenario 2) Operations Costs by Year

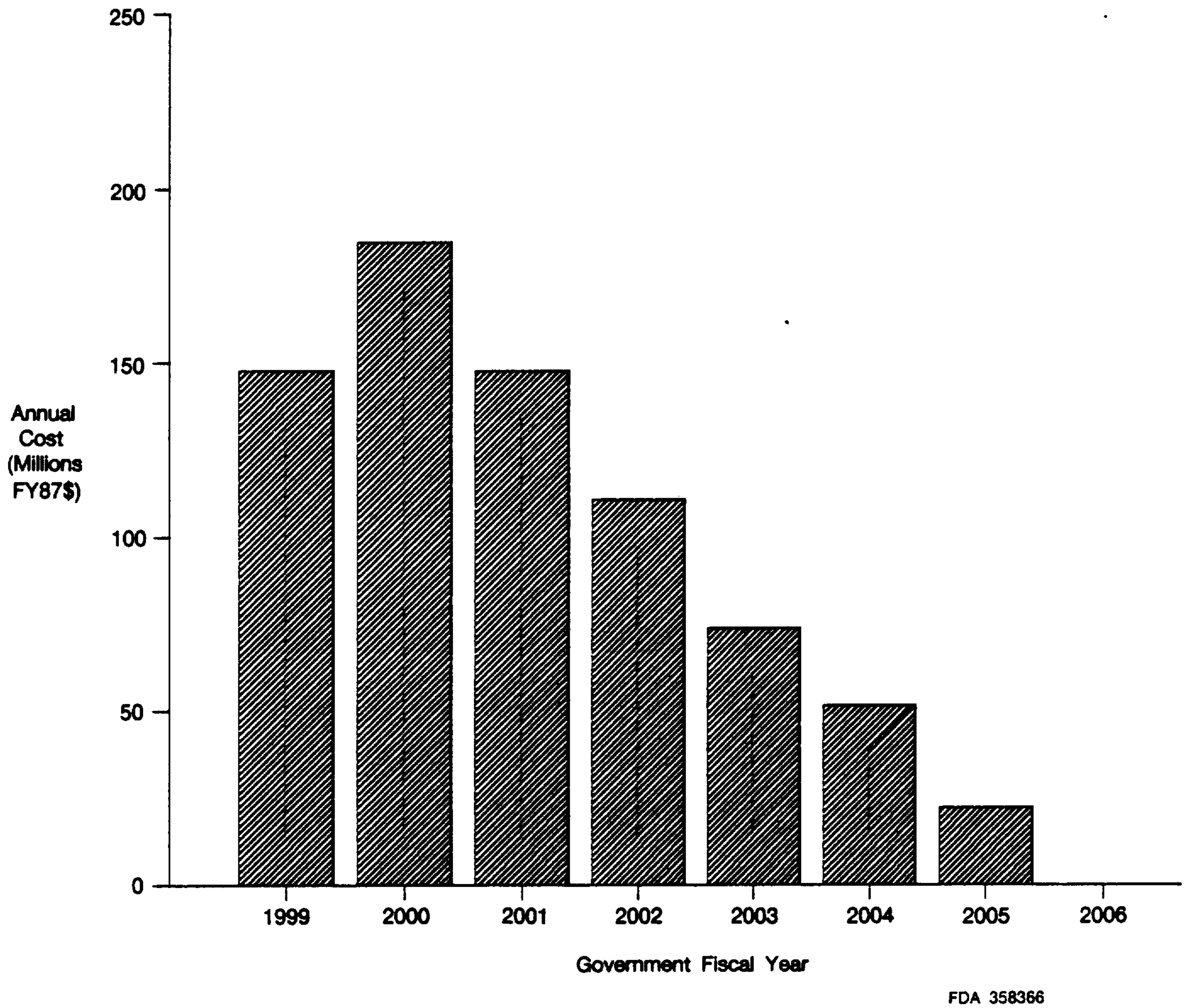

Figure 5-9. STME/Derivative STBE (Scenario 2) Product Improvement and Support Program Costs by Year

Table 5-8. *STME and Derivative STBE Recurring Production Theoretical First Unit Costs*

System	*STME TFU (FY87$)*	*Derivative STBE TFU (FY87$)*	*Derivative STBE Cost Commonality % STME TFU*
STBE Hardware	11349K	10305K	72%*
Turbomachinery	2867	3045	58
HPOTP	1379	1445	35
HPFTP	1488	1600	80
Combustion Devices	4046	2595	77*
Main Injector	330	330	100
Thrust Chamber	585	655	0
Nozzle	961	961	100
Nozzle Skirt	1521	—	—
Gas Generator	357	357	100
Igniters	292	292	100
Controls	1544	1644	68
Controllers/Monitors/Software	506	506	95
Sensors	285	285	100
Valves/Actuators	670	770	30
Interconnects	83	83	100
Propellant Feed	1686	1780	84
Ducts	939	1033	80
Miscellaneous (System Hardware)	747	747	90
Support Devices	663	698	65
Gimbal	235	270	0
Tank Repressurization	261	261	100
Start System	17	17	100
POGO Flight System	150	150	100
Integration, Assembly & Test	143	143	100
Acceptance Test	400	400	100

* Reflects % of applicable STME hardware costs.
Notes: 1. All costs in thousands of FY87 dollars.
2. Lot size = 100.

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Table 5-9. *STME and Derivative STBE Recurring Operations Unit Cost*

	Theoretical First Unit		*100th Mission, 10 Missions/yr*	
	STME	*Derivative STBE*	*STME*	*Derivative STBE*
Program Management	104.7	104.7	7.2	7.2
System Engineering and Integration	401.1	401.1	25.3	25.3
Facilities Maintenance	0	0	0	0
Operations and Support				
Launch Operations	12.6	12.6	5.0	5.0
Flight Operations	170.3	170.3	11.7	11.7
Spares Replenishment	59.8	65.8	23.7	26.1
Recovery Operations	10.8	10.8	4.3	4.3
Refurbishment Operations	153.9	153.9	61.0	61.0
Training	5.0	5.0	2.0	2.0
Total Operations Cost, $/Engine/Mission	918.3	924.3	140.2	142.6

Note: All costs are in thousands of FY87 dollars.

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Table 5-10 compares Design and Development costs for the STME portion of the STME/Derivative STBE program with similar costs for the Baseline STME program (See FR-19830-2). The STME costs in the derivative program are $215M less than for the baseline STME program because the engine is used only on the core stage. Total Design and Development costs for the STME/Derivative STBE program including the STBE are $443M more ($1841M vs. $1348M) than for the Baseline STME program.

Table 5-10. *Gas Generator STME/Derivative STBE Program — STME Design and Development and Cost Comparison*

	*Baseline STME Core & Booster**	*STME Core Only*	*STME Cost Difference*
Program Management	$70M	$66M	$ 4M
System Engineering and Integration	60	42	18
Engine Design and Development	180	171	9
Engine Test			
Test Hardware	329	352	-23
Test Operations and Support	246	254	-8
Flight Test Hardware	208	73	135
MPTA Test Hardware	100	37	63
Facilities			
Production	8	8	0
Launch	4	4	0
Test	22	22	0
Software Engineering	13	12	1
GSE	26	19	7
Tooling	68	68	0
Special Test Equipment (STE)	25	25	0
Operations and Support	39	30	9
Total DDT&E Program Cost	$1,398M	$1,183M	$215M

* Baseline Gas Generator STME DDT&E program costs for Scenario 1 reported in FR-19830-2.
Note: All costs in millions of FY87 dollars.

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The program cost estimates generated in this study indicate that P&W's STME/Derivative STBE design will result in a low-cost engine program. The low recurring engine costs for the STME/Derivative STBE should permit the ALS program to achieve its objective of significantly reducing the cost of placing large payloads into orbit.

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